N71-19241
NASA CR-116893

Report No. A-9

FINAL REPORT (NASW-2023)

Report No. A-9

FINAL REPORT (NASW-2023)

by

Astro Sciences

of

IIT Research Institute

Chicago, Illinois

for

Lunar and Planetary Programs
Office of Space Science and Applications
NASA Headquarters
Washington, D. C.

Seventh Annual Summary Report

APPROVED:

C. A. Stone

C. A. Stone, Director
Physics Research Division

November 1970

FOREWORD

This final report summarizes the reports prepared and the special tasks performed by Astro Sciences of IIT Research Institute during the twelve month period from November, 1969 through October, 1970. Eleven reports or technical memoranda are summarized together with a description of two memos on which no formal reports have been written. In addition the abstracts of four technical papers, which will be published in the open literature, have been included. This work has been performed under NASA Contract Number NASW-2023.

TABLE OF CONTENTS

FINAL REPORT (NASW-2023)

LONG RANGE PLANNING FOR SOLAR SYSTEM EXPLORATION

NOVEMBER, 1969 - OCTOBER, 1970

1. INTRODUCTION

Astro Sciences of IIT Research Institute (AS/IITRI) has been engaged in a program of advanced research, study and analysis for the Planetary Programs Division (Code SL) of NASA since March, 1963. The results of Astro Sciences' work up to October 31, 1969, have been previously reported[1]. This report summarizes the work performed on Contract NASW-2023 from November 1 through October 31, 1970.

The purpose of advanced mission planning is to derive a preliminary understanding of those missions, and associated mission requirements, which are of importance in the evolution of knowledge of our solar system. It is necessary not only to have a solid foundation in science and engineering for this type of planning but also the ability to integrate the increasing awareness of the problems involved in space exploration back into the advanced planning process. Astro Sciences' program

---

[1] The contract work conducted between March 1, 1963 and December 1, 1968 summarized in AS/IITRI Report No. A-6, "Long Range Planning Studies for Solar System Exploration" (1969). Work done between December 1, 1968 and October 31, 1969 is summarized in AS/IITRI Report No. A-7, "Long Range Planning for Solar System Exploration" (1970).

during the period covered by this report, as it has during the previous six years, has continued to develop this process in accordance with NASA's broadening needs for advanced planning.

The continuing activities of Astro Sciences are reported to the Planetary Programs Division at regularly scheduled bi-monthly review meetings. However, the most tangible output is in the form of technical reports and memoranda. During the twelve months covered by this report a total of eleven reports or technical memoranda have been submitted. Summaries of these documents are given in Section 2. Section 3, Special Studies and Technical Notes, summarizes study efforts that have been performed and capabilities that exist but for which no formal reports have been published. Section 4 contains abstracts of those papers published and presented by Astro Sciences staff members which originated primarily as a result of work performed under this contract. Section 5 contains a bibliography of reports and technical memoranda published by AS/IITRI. Finally Section 6 summarizes the major computer programs used to support Astro Sciences' technical efforts.

2. SUMMARY OF REPORTS AND TECHNICAL MEMORANDA PUBLISHED NOVEMBER, 1969-OCTOBER, 1970

2.1 MISSION OBJECTIVES

Report No. P-36
"THE PHYSICAL STRUCTURE OF THE SATURN RING SYSTEM"
M. J. Price
November 1970.

Planetary mission analysts are currently concerned by the potential hazard the Saturn ring system presents to spacecraft navigating near the planet. Uncertain knowledge of the physical properties of the ring system has caused a conservative attitude to prevail; for safety fly-by trajectories have been constrained to pass well outside the rings. Since any increase in our knowledge of the ring properties would broaden the range of practical options available to Grand-Tour missions, further study has been made of the available ground-based photometry of the system.

In a critique of earlier studies of the physical structure of the ring system, the relevance of each type of ground-based photometry is discussed. Of the available photometric data, the most useful for deriving a physical model for the ring system are measurements of the surface brightness as a function of radial distance from Saturn, solar illumination angle, and phase angle. Using these data, together with the Mie theory of scattering, a new, optically self-consistent ring model has been developed. The ring particles appear to be ice crystals of characteristic radius $\sim 0.1\mu$. In the densest region of the ring system (ring B), the surface density (i.e., the fraction of the ring plane covered by particles in a perpendicular projection) is 0.4 ± 0.2. In the analysis, the geometrical thickness of the ring system remains arbitrary. In addition, because of insufficient observational data, no information has been obtained on the size distribution of the ring particles.

Caution must be exercised in estimating the implied navigational hazard since the ring model cannot be considered final. Even so, the probable effect on a TOPS-class spacecraft, intersecting the densest region of the ring system, while on a 4-planet (J-S-U-N) Grand-Tour mission, has been studied. The trajectory of the spacecraft intersects the ring plane at an angle of 60° to the normal; the spacecraft velocity relative to Saturn is then 30 km/sec in the posigrade direction. Individually, particle impacts should cause negligible damage, since the TOPS design calls for the spacecraft to withstand impacts by particles of mass less than 10^{-2}g striking at 20 km/sec; the ring particles are of mass $\sim 10^{-15}$g, while the relative spacecraft particle velocity is $\sim$ 10 km/sec. Collectively the ring particles will reduce the spacecraft velocity by $\sim$ 0.2 m/sec through interchange of momentum. On the basis of current knowledge the ring system does not represent a significant hazard for Grand-Tour missions.

2.2 MISSION ANALYSIS

Report No. M-20
"JUPITER ORBITER MISSION STUDY"
Compiled by J. C. Niehoff
September 1970.

Pioneers F and G will be launched in 1972 and 1973, respectively, on flyby missions to Jupiter. Current program plans strongly suggest that these missions be followed by dual launches of two Grand Tour missions: a Jupiter-Saturn-Pluto Tour in 1977 and a Jupiter-Uranus-Neptune Tour in 1979. Altogether these missions constitute a launch endeavor of six flyby spacecraft to Jupiter in the 1970's. Clearly, now is the time to move ahead with advanced planning and analysis of more comprehensive Jupiter orbiter and atmospheric probe missions which, hopefully, will closely follow or perhaps even mesh with the Grand Tour missions. The purpose of this study is to identify first-generation Jupiter orbiter missions, evaluate their requirements, and determine their contribution to Jupiter exploration.

For this first look at Jupiter orbiter missions it was necessary to specify several broad study guidelines which would identify and constrain the scope of analysis. The guidelines used are as follows:

- All aspects of Jupiter, its satellites and the Jovian electromagnetic environment be assessed in determining the relevant science objectives for orbiter missions,

- The mission flight mode be restricted to ballistic transfers emphasizing opportunities during the period 1974-85,
- Mission configurations and payloads be identified which are compatible with Titan launch vehicles.

The analysis of Jupiter orbiter missions was divided into four areas of consideration: 1) definition and evaluation of science objectives, 2) measurement specification development and parameterized instrument design, 3) trajectory analysis and orbit selection, and 4) construction and comparison of mission alternatives. It is appropriate to summarize the study results in terms of these study areas.

Science definition and evaluation for first-generation Jupiter orbiters were accomplished by a systematic breakdown of the broad goal of Jupiter exploration into successive levels of detail. The goal of Jupiter exploration was first broken down into regimes closely aligned to the various natural characteristics of the planet, e.g., interior, atmosphere, surrounding magnetic field and biology. Each of these regimes has in turn specific categories of interest. In the case of atmosphere there are the topics of composition, structure and dynamics. Continuing this process of deduction and refinement each category was parceled into objectives which, in total, provide a complete definition of the category. For example, atmospheric composition really means identifying the objectives; 1) elemental and molecular abundances, 2) isotopic abundances and ratios and 3) particulate matter. The final step to the process was identifying the measurables which are the physical evidence of each objective.

A summary of the categories, objectives and measurables which were specifically relevant to orbiter missions of

Jupiter, are presented in the left-hand column* of the instrument selection chart presented in Figure 2-1. The majority of measurables identified in Figure 2-1 deal with the atmosphere and particle/field exploration regimes. Few measurables dealing with Jupiter's surface and interior, or biology, were considered suitable for remote sensing techniques.

A numerical percentage evaluation of the identified measurables indicated that they constitute approximately 30% of the goal of total Jupiter exploration. Although such an evaluation is subjective in nature, and must be continually revised as new knowledge of the planet is acquired, it does provide added confidence in the value of the orbiter mission mode to Jupiter exploration.

Measurables related to the Jovian satellites are not included in Figure 2-1. They can be summarized, however, as preliminary in character appropriate to satellite flybys by an orbiting Jupiter spacecraft. Regional scale imagery of surface features, detection of an atmosphere, measurement of an inherent magnetic field and the interaction of the satellite with Jupiter's magnetosphere, and evaluation of the satellite's physical properties (mass, diameter, rotation rate and oblateness) are all experiments of significance for initial satellite investigation.

The second study area dealt with the question: "How can the defined measurables be investigated?" To answer this question measurement specifications were written for measurement techniques and instrument types identified with each measurable. The particular techniques and instruments selected for consideration are summarized in the remainder of Figure 2-1.

---

* The remainder of the chart will be described in the summary of measurement specifications and instrument design which follows.

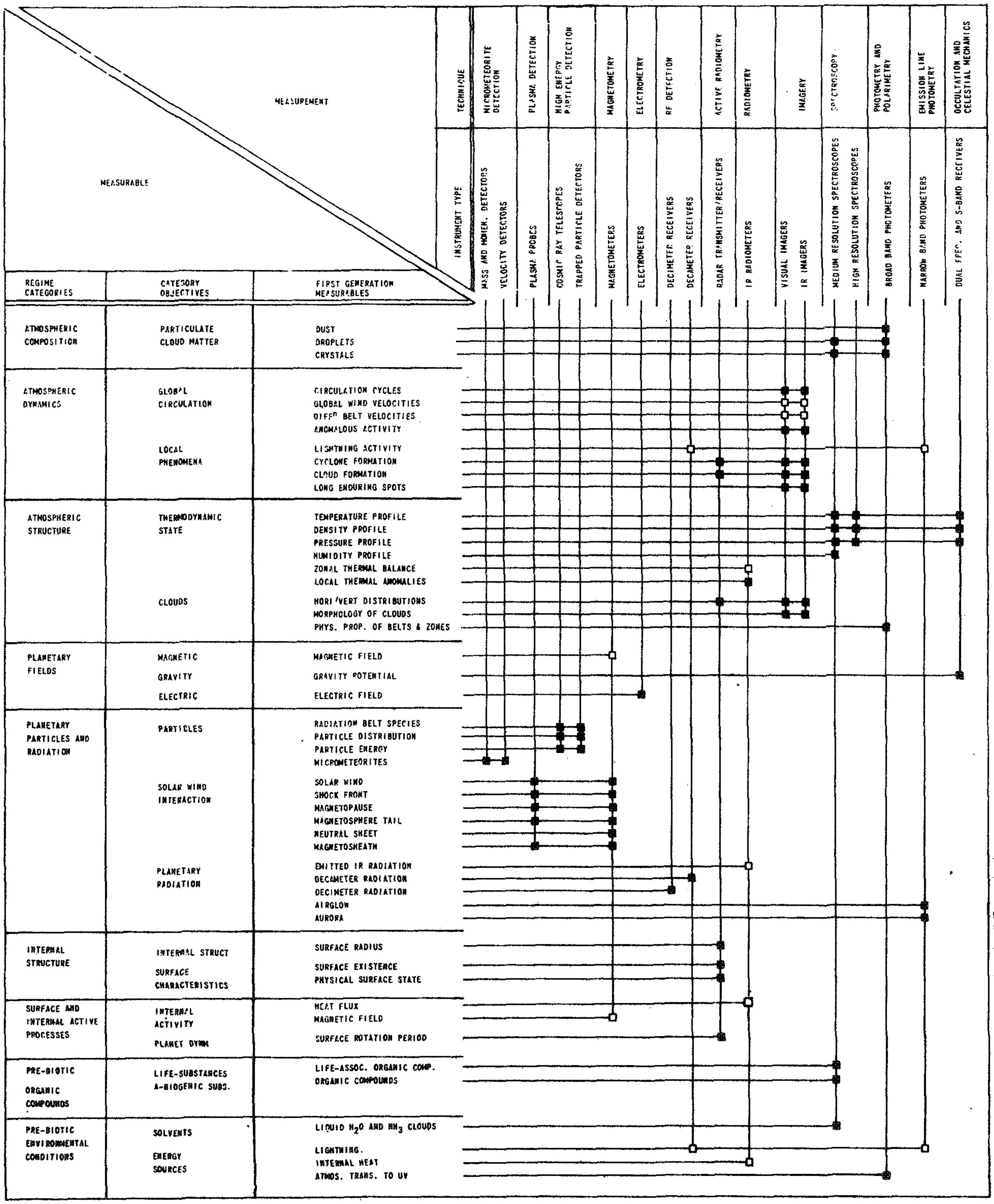


FIGURE 2-1. MEASUREMENTS AND INSTRUMENTS

A total of 18 different instrument types are of relevance to the 50 measurables defined. A closer look at the instrument types revealed that they all fit into one of two instrument classes: 1) in situ particle/field detectors, and 2) remote planetary sensors. The open squares in Figure 2-1 indicate measurables which are duplicated under several category objectives.

The measurement specifications developed for each measurable-instrument combination included estimates of desired wavelength energy range, pass bands, spectral and spatial resolution, coverage, distribution, acquisition and repetition time, solar illumination, positional accuracy, and prior measurement requirements. From these data conceptual instruments were selected (or designed) which could provide acceptable measurable information.

A summary of weight, power and data rate of particle/field instruments selected for consideration on Jupiter Orbiter Missions is presented in Table 2-1. A similar summary of key planetology instruments is given in Table 2-2. The particle/field instruments are acceptable designs largely borrowed from previous spaceflight projects. The planetology experiments (primarily imagers), on the other hand, are conceptual designs developed to match the appropriate measurement specifications. In general, they are advanced state-of-the-art designs characterized by high sensitivity and spatial resolution which are necessary for measurements from an orbiting Jupiter platform. The instruments listed in Tables 2-1 and 2-2 constitute the basic "shopping list" from which science payloads were later evolved in the study.

The third study area was an analysis of the energy requirements for ballistic Jupiter orbiter missions. From the results an assessment of payload capability of Titan launch systems was possible and candidate orbit sizes were determined. The trajectory characteristics of an 11-year cycle of annual

TABLE 2-1

SUMMARY OF JUPITER PARTICLE AND FIELD INSTRUMENTS

TECHNIQUE	INSTRUMENT	RANGE	WEIGHT	POWER	DATA RATE
MAGNETOMETRY	FLUXGATE MAGNETOMETER	$0.1\ \gamma - 100$ G	7 LBS.	3 W	15 BPS
	VECTOR HE MAGNETOMETER[1]	$0.1\ \gamma - 10$ G	10	2.5	15
PLASMA DETECTION	ELECTROSTATIC ANALYZERS[1]	E: 1-500 EV P: 0.1-8 KEV	10	4	50
	FARADAY CUP	E: 3-300 EV P: 0.12-5 KEV	8	3	48
HIGH-ENERGY PARTICLE DETECTORS		E: 0.05-50 MEV P: 0.3-500 MEV	3-6	1.5-3	50
ELECTRIC FIELD MEASUREMENT	IMP-I	0-300 HZ	4[2]	1.5	15
RF DETECTION	DIRECTIONAL ANTENNA	300 KHZ-300 GHZ	4[2]	3	15
MICROMETEOROID DETECTION	"SISYPHUS"[1]	SIZE: $>50\ \mu$M VEL: 0.01-70 KM/S	5	2	1
	PRESSURE CELLS[1]	$>10^{-9}$ GM	6	0.1	1

1. PIONEER F/G INSTRUMENTS

2. DOES NOT INCLUDE ANTENNA ($\sim$4 LBS).

TABLE 2-2

PLANETOLOGY INSTRUMENT SUMMARY

INSTRUMENT	ALTITUDE RANGE (JUPITER RADII)	RESOLUTION RANGE (KM)	WEIGHT (LB)	POWER (WATT)	MAXIMUM DATA RATE
RETURN BEAM VIDICON	2-50	3.5-100	35	26	100K BPS
UV SPECTRO-PHOTOMETER	2-5	400-1000	25	4	100 BPS
IR RADIOMETER	2-8	34-100	28	5	2K BPS
3-CHANNEL IR RADIOMETER	2-5	95-200	78	5	700 BPS
UV PHOTOMETER	2-5	40-100	18	2	0.2 BPS

launch opportunities were studied for the period 1974-1985. The trajectory analysis was summarized into three transfer classes: 1) the lowest average energy transfers with 10-day launch windows, flight time $\approx$ 760 days, 2) the lowest average energy transfers with 20-day launch windows, flight time $\approx$ 710 days, and 3) short 500-day transfers with 20-day launch windows.

The three transfer classes were combined with four Titan launch vehicle configurations to form five flight modes which are presented in Table 2-3. Energy and gross payload data are also given. These modes were selected (from a total of 12 possible combinations in Table 2-3) such that launch vehicle size and flight time could be traded off with injected payload being kept fairly constant. Ideally this selection rationale would amount to moving diagonally through the matrix of transfers versus launch vehicles from the upper left-hand corner to lower right-hand corner. The selected flight modes were both, 1) useful to the selection of appropriate orbit sizes and 2) necessary for determination of payload feasibility of mission alternatives developed in the fourth study area.

To limit the amount of orbit analysis associated with the definition of mission alternatives, a set of four orbit sizes was selected to represent the range of interest for orbiter missions. A short analysis of Jupiter radiation hazards indicated that an orbit periapse radius (r_p) of three Jupiter radii (R_J) was necessary to insure long orbital staytimes ($\sim$ 1 year). Using a fixed r_p - 3 R_J, the four selected orbits are presented in Figure 2-2. Starting with the smallest orbit, their periods are 7.5, 15, 30 and 45 days long, respectively. The 7.5-day orbit represents a lower limit on orbit size due to minimum payload considerations while the 45-day orbit represents the maximum size orbit necessary to cover the entire Jupiter environment. Properly oriented, the apoapse (r_a = 98 R_J) of the

TABLE 2-3

SELECTED ORBITER DELIVERY MODES

TRANSFER CLASS ▽ / LAUNCH VEHICLE ▷	TITAN III D CENTAUR	TITAN III D CENTAUR BURNER II	TITAN III F CENTAUR	TITAN III F CENTAUR BURNER II
CLASS ONE FLIGHT TIME = 760^d LAUNCH WINDOW = 10^d	MODE ONE V_c = 48000 FT/SEC W_{inj} = 1950 LBS. VHP = 7.25 KM/SEC	MODE TWO V_c = 48000 FT/SEC W_{inj} = 2550 LBS. VHP = 7.25 KM/SEC	MODE THREE V_c = 48000 FT/SEC W_{inj} = 3300 LBS. VHP = 7.25 KM/SEC	
CLASS TWO FLIGHT TIME = 710^d LAUNCH WINDOW = 20^d			MODE FOUR V_c = 49000 FT/SEC W_{inj} = 2700 LBS. VHP = 8.00 KM/SEC	
CLASS THREE FLIGHT TIME = 500^d LAUNCH WINDOW = 20^d				MODE FIVE V_c = 52000 FT/SEC W_{inj} = 1950 LBS. VHP = 13.50 KM/SEC

ORBIT PERIOD, DAYS	CAPTURE IMPULSE, KM/SEC.			RADIATION LIFETIME	
	MODES 1,2,3	MODE 4	MODE 5	REVS	DAYS
7.5	2.465	2.628	4.264	425	3188
15	1.822	1.985	3.621	450	6750
30	1.424	1.586	3.222	470	14100
45	1.264	1.427	3.063	475	21375

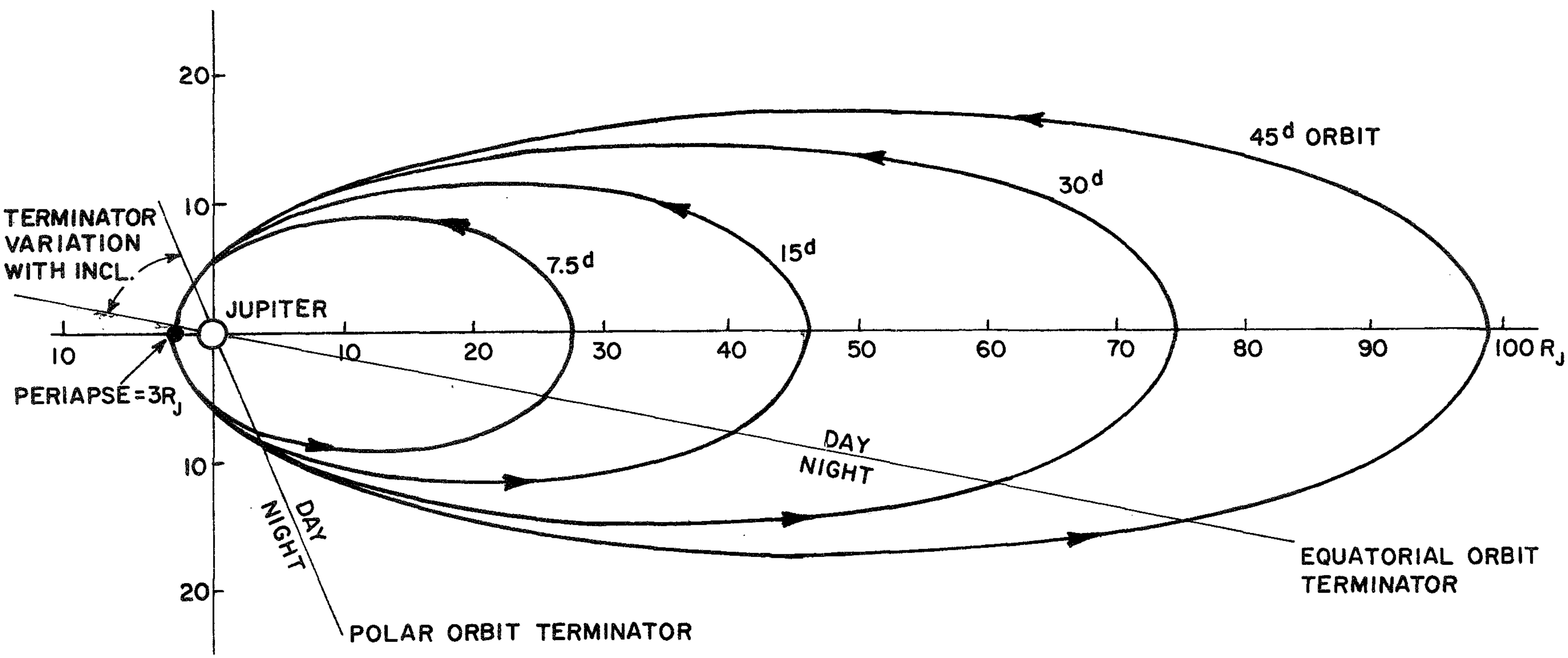


FIGURE 2-2. REPRESENTATIVE JUPITER ORBITS

45-day orbit would lie outside the solar-wind shock front of the Jovian magnetosphere estimated to lie 50-65 R_J from the planet. Note that minimum capture impulses for each flight mode and radiation lifetimes are also given in Figure 2-2 for each orbit.

The development of mission definitions (fourth study area) was a synthesis of results obtained in the first three study tasks. It quickly became apparent, in developing these definitions, that no single orbit would satisfy all science objective and instrument requirements. In fact, two opposing trends were implied by the requirements, short-period orbits versus long-period orbits. The advantages of small orbits (short periods) pertain to measurements of the planet itself. For altitude (resolution)-limited planetology instruments, the short-period orbits provide more opportunities to image the planet over a fixed period of time. Advantages of long-period orbits pertain to, 1) particle and field measurements and their spatial requirements, and 2) spacecraft design including weight and radiation lifetime.

The split in orbit capabilities suggests a logical division of science objectives for Jupiter orbiter missions, 1) particle and field measurements, and 2) planetology measurements. Recall that a similar split was found in the development of candidate instruments for Jupiter orbiter missions. Four different mission profiles were suggested to investigate the trade-offs of this situation. These are given in Table 2-4. Mission No. 1 is specifically a particle and field mission. It involves two spacecraft in 45-day orbits at two inclinations. The multiple spacecraft and different orbit inclinations are selected to improve the spatial coverage of the enormous region of the Jovian magnetosphere. Mission No. 2 is specifically a planetology mission using a 15-day inclined orbit to maximize

TABLE 2-4

JUPITER ORBITER MISSION SELECTIONS

MISSION NUMBER	SCIENCE OBJECTIVE	ORBIT PARAMETERS		
		PERIAPSE (R_J)	PERIOD (DAYS)	INCLINATION (DEG)
1	PARTICLES AND FIELDS[1]	3	45	0 AND 120[1]
2	PLANETOLOGY	3	15	60
3	PLANETOLOGY AND SATELLITE OBSERVATIONS	2.391 (2.290)[2]	7.111 (14.222)[2]	0
4	PARTICLE AND FIELD AND PLANETOLOGY	3	30	60

1. TWO SPACECRAFT ORBITED AT DIFFERENT INCLINATIONS
2. NEXT ORBIT CHOICE IF PREFERRED ORBIT CANNOT BE ACHIEVED

the coverage of altitude-limited instruments. Mission No. 3 also a planetology mission, reduces the latitude coverage by using an equatorial orbit in exchange for multiple opportunities to observe the Galilean satellites*. Mission No. 4 is designed to do both particle and field and planetology experiments. Its orbit period of 30 days is definitely a compromise between the opposing measurement requirements on orbit size.

A breakdown of the selected science packages and required spacecraft subsystem weights for each of these four mission definitions is presented in Table 2-5. Mission No. 1 (Particle and Fields) uses a spin-stabilized spacecraft which could be a modified Pioneer F/G design. The total spacecraft weight is small enough to consider a double spacecraft launch with a single Titan launch vehicle.

Mission No. 2 (Planetology) only carries planet-oriented experiments. Emphasis is therefore placed on tight orbits (~ 15-day period) to improve the resolution/coverage characteristics of the experiments. Mission No. 3 science (Planetology and Satellites) is similar to that of Mission No.2, except several particle and field instruments were added to measure magnetospheric disturbances of the Jovian satellites. Two instruments, (narrow band photometer and X-ray imager), which are not as useful in equatorial satellite observation orbits, have been dropped leaving the total science payload unchanged. Again tighter orbits are desirable.

Mission No. 4 is a compromise which automatically places its orbit period in the 30-day (one month) range. Its

---

* This technique is discussed in detail in AIAA Paper Number 70-1070, "Touring the Galilean Satellites", presented by J. Niehoff at the AAS-AIAA Astrodynamics Conference, August 1970. An abstract of this paper can be found on page 113 of this report.

TABLE 2-5

SCIENCE AND SPACECRAFT WEIGHT BREAKDOWNS

	MISSION 1	MISSION 2	MISSION 3	MISSION 4
SCIENCE INSTRUMENTS:				
VECTOR HELIUM MAGNETOMETER	10 lb.		10 lb.	10 lb.
ELECTROSTATIC PLASMA ANALYZERS(2)	9			
LEPEDEA	4			4
PARTICLE DETECTOR AND DOSIMETER	6			6
GEIGER TUBE TELESCOPE	3		3	3
TRAPPED RADIATION DETECTORS(4)	5		5	5
SWEPT FREQUENCY RF RECEIVER	8		8	8
DC ELECTRIC FIELD DETECTOR*	4			4
MICROMETEORITE DETECTORS(2)	11	11	11	11
RETURN BEAM VIDICON		35	35	35
NEAR-IR LINE SCANNER		28	28	28
THREE-CHANNEL IR RADIOMETER**		28	28	28
UV SPECTROPHOTOMETER		25	25	25
NARROW BAND UV PHOTOMETER		18		18
X-RAY IMAGER		8		8
IONOSONDE		25	25	25
TOTALS	60 lb.	178 lb.	178 lb.	218 lb.
SPACECRAFT:				
SCIENCE	60 lb.	178 lb.	178 lb.	218 lb.
IR RADIOMETER COOLING SYSTEM		50	50	50
COMMAND CONTROL AND SEQUENCING	17	45	50	50
DATA MANAGEMENT AND STORAGE	35	105	105	105
TELEMETRY (INCLUDES ANTENNA)	43	165	165	165
RTG POWER AND CONDITIONING	150	275	275	300
THERMAL CONTROL	10	25	25	25
ALTITUDE CONTROL	40	150	150	150
GUIDANCE SENSOR SYSTEM		30	30	30
PLANETOLOGY INSTRUMENT PLATFORM		35	35	35
STRUCTURE (INCL.BOOMS) AND SHIELDING	75	175	205	235
CONTINGENCIES (~ 15%)	70	217	232	237
TOTAL	500 lb.	1450 lb.	1500 lb.	1600 lb.

* USES RF RECEIVER ANTENNA

** REQUIRES AN ADDITIONAL 50 lb. SOLID METHANE COOLING SYSTEM.

spacecraft is the heaviest of the four, carrying a complete complement of science instruments. Its orbit, however, is not as good (i.e., too large) as those of Mission Nos. 2 and 3 from the standpoints of planetology resolution and coverage. Yet, for its particle and field measurements, the orbit could be even larger. Its heavier spacecraft weight also suggests a preference for larger orbits. This mission was not judged to be a good alternative.

The total injected payload from earth required by each mission is compared with the payload capability of each selected flight mode in Figure 2-3. Mode 5 is not included since its energy characteristics were so high that its small payload capability was obviously irrevelant. Mission No. 1, Particle and Fields Orbiter, is divided into two options (1A and 1B) in the figure. Option 1A is the payload required for launch of a single spacecraft while Option 1B is the payload required for two spacecraft on one launch vehicle. The flight modes (launch vehicle-transfer combinations) were defined in Table 2-3. It is concluded that a Titan III F/Centaur launch vehicle and probably a space-storable propulsion system will be required to perform any of the suggested planetology missions. The Titan III D/ Centaur is more than adequate for Mission No. 1A and is also acceptable for Mission No. 1B if space-storable propulsion is used. In general, a 760-day transfer with a 10-day launch window will be required. However, during the low-energy opportunity series of 1974-1976 and 1980-1984 the launch window can be enlarged to more practical values of 15-20 days. The consideration of high-energy upper stages, e.g., a hydrogen/ fluorine kick stage or a solar-electric low-thrust stage, on the Titan vehicles has not been considered. Such additions should be studied as alternatives to seven-segment solids for the Titan and development of space-storable retro systems.

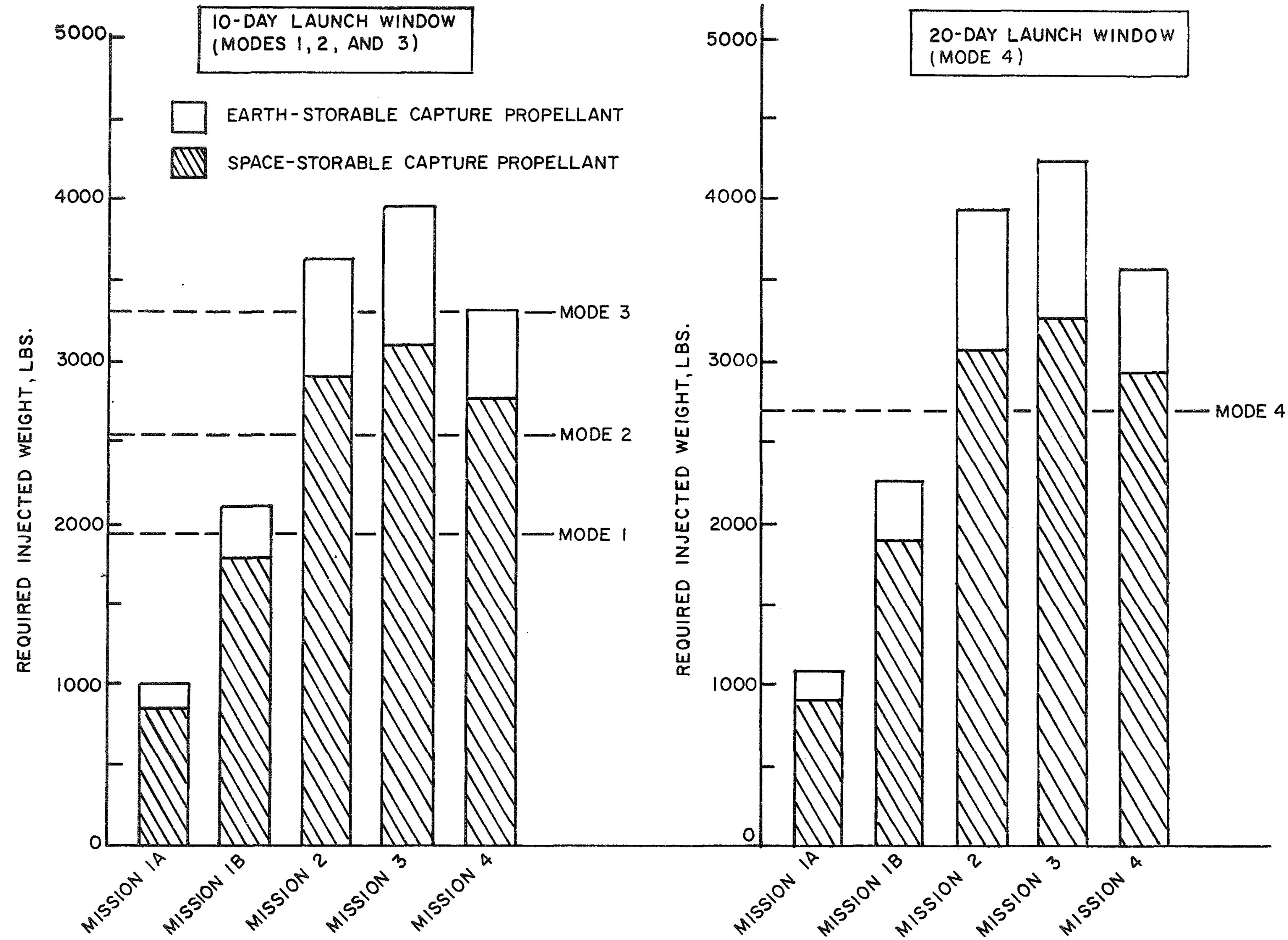


FIGURE 2-3. COMPARISON OF PAYLOAD REQUIREMENTS WITH FLIGHT MODE CAPABILITIES.

Four specific recommendations are made as conclusions to the study:

1) Press for a double Pioneer P/F orbiter mission launched by a single Titan III D/Centaur/Burner II during the series of low-energy Jupiter opportunities from 1974 to 1976.

2) Encourage the continued development of better planetology instruments to cope with the resolution/intensity problems at Jupiter.

3) Plan to initiate planetology orbiter missions after the Grand Tour missions during the series of low-energy Jupiter opportunities from 1980 to 1984.

4) Continue with plans for early atmospheric probe missions since orbital instruments will not provide data below Jupiter's cloud tops.

Report No. M-25

"A PRELIMINARY STUDY OF COMPOSITE ORBITER/LANDER MISSIONS TO THE SATELLITES OF THE OUTER PLANETS"

M. J. Price and D. J. Spadoni

September 1970

A preliminary study has been made of the scientific objectives and payload requirements for landing unmanned spacecraft on satellites of the four giant outer planets. Scientific and operational rationale are developed for selecting six major satellites for composite orbiter/lander missions. Specific missions to Io (Jupiter I), Europa (Jupiter II), Ganymede (Jupiter III), Callisto (Jupiter IV), Titan (Saturn VI) and Triton (Neptune I) are considered. Two classes of lander missions (of equal mass in satellite orbit) are discussed, 1) a single soft-lander, and 2) multiple (10) rough-landers.

The major objective of such missions would be the collection of scientific data pertinent to:

1) a better understanding of the mode(s) of formation of the satellites and smaller planets of the solar system,

2) the study of the origin of planetary/satellite systems,

3) comparing theories for the evolution of planet/satellite systems with those for the evolution of the solar system itself.

Lander experiments should emphasize identification of fundamental chemical and physical properties of the satellite. The orbiting bus, regarded as an essential communication link between the lander and earth, could enhance these measurements

by generating global surface feature and thermal maps through orbital imagery. With these data the satellites could be compared with the smaller terrestrial planets and the moon, hopefully providing new insight into the vast differences of the inner and outer planets. Surface experiments and instrumentation appropriate for initial landings are briefly discussed.

A second objective for satellite lander missions is the use of these bodies as bases for the remote observation of their parent planets. A satellite base has the inherent advantage of platform stability compared with an orbiting spacecraft. Also, if the satellites' rotation periods are locked to their orbital periods (e.g. the moon and earth), as is predicted, then the parent planet is continuously observable from any landing site on the "front-face" of the satellite. Since the six satellites selected all apparently revolve well outside the intense regions of planetary radiation belts, radiation hazards should not be a major concern. The constant altitude of these regular satellites above their parent planets would simplify imagery requirements for planetary observations.

The satellites were also considered as bases for monitoring the magnetospheres surrounding the parent planet. However, it was concluded that, at least until the existence and characteristics of outer planet magnetospheres have been better established, the disruptive effects of the satellites presence would make such measurements difficult to interpret. Hence a payload consisting solely of particle and field instruments was considered inappropriate for any lander mission.

Trajectory and payload analyses were performed for the composite orbiter/lander mission to each of the six regular satellites identified above. The class of soft-lander missions were sized to a useful landed payload of 1000 lbs., exclusive of terminal guidance (descent radar), variable-thrust propulsion and landing gear. Experiment instrumentation was limited to 100 lbs. For the class of rough-lander missions 60 lbs. useful payload (exclusive of the impact limiter) was allowed at impacts $\leq$ 200 ft/sec. The associated science was limited to 10-15 lbs. In either case the supporting orbiter was assumed to weigh 1500 lbs.

Payload requirements were determined by separating the mission into four distinct phases and applying various propulsion systems to each phase. The phase breakdown, in reverse order of occurrence, is as follows:

1) Terminal landing maneuver; variable-thrust (earth-storable) chemical propulsion considered for soft-lander, free-fall assumed for rough-lander.

2) Deorbit and braking maneuvers; chemical* propulsion considered for both deorbit impulse and constant-thrust braking maneuver just prior to terminal descent.

3) Polar satellite orbit insertion; chemical* three-impulse maneuver sequence from planet approach of ballistic and solar-electric low-thrust interplanetary transfers, spiral low-thrust approach from nuclear-electric transfer followed by single-impulse chemical propulsion capture maneuver.

4) Interplanetary transfer; ballistic, solar-electric low-thrust, and nuclear-electric low-thrust flight modes considered.

---

* Candidate chemical propellants include earth-storable, solid, space-storable and cryogenic.

Payload results indicated that a nominal total useful weight of 4000 lbs. was required in a 100-km polar circular satellite orbit to perform the defined soft-lander missions (this includes the 1500 lbs. communications relay and mapping orbiter). In order to apply the interplanetary trajectory and payload analyses equally to each mission class, the rough-lander missions were also permitted a total useful in-orbit weight of 4000 lbs. Analysis of the rough-lander propulsion requirements showed that ten landers, their support carriage and the orbiter were within this weight allowance.

Assuming that combinations of earth-storable, space-storable, cryogenic and solid propulsion systems can be made available for satellite capture, deorbit, braking and landing maneuvers, the payload feasibility of either lander-class mission can be summarized in terms of the interplanetary flight mode employed. This is done in Table 2-6.

Missions using ballistic interplanetary trajectories are conceptually possible to all six selected satellites using Saturn-class launch vehicles. A mission to Callisto is feasible with the Intermediate-20/Centaur if cryogenic propulsion is used for the capture and braking maneuvers. The Saturn V provides mission capability to Europa, Ganymede and Callisto without regard to the type of propulsion used at the satellite. Adding a Centaur to the Saturn V makes possible missions to all four Galilean satellites of Jupiter and the more distant satellites, Titan (Saturn) and Triton (Neptune), with flight times ranging from about 2 years to the Galilean satellites to 11 years to Triton.

Solar-electric low-thrust missions are possible to Europa, Ganymede, Callisto and Titan with the Intermediate-20/ Centaur as a launch vehicle. The flight times are comparable

TABLE 2-6

MISSION SUMMARY

INTERPLANETARY TRANSFER MODE LAUNCH VEHICLE COMBINATION	IO	EUROPA	GANYMEDE	CALLISTO	TITAN	TRITON
BALLISTIC:						
Intermediate-20/Centaur	No	No	No	$600\text{-}700^{d}$	No	No
Saturn V	No	$600\text{-}700^{d}$	$500\text{-}700^{d}$	$500\text{-}600^{d}$	No	No
Saturn V/Centaur	$600\text{-}700^{d}$	$500\text{-}700^{d}$	$500\text{-}700^{d}$	$500\text{-}700^{d}$	$3\frac{1}{2}\text{-}4^{y}$	$11\text{-}12^{y}$
SOLAR-ELECTRIC:						
Intermediate-20/Centaur	No	$900\text{-}1100^{d}$	$900\text{-}1000^{d}$	$600\text{-}700^{d}$	$4\text{-}4\frac{1}{2}^{y}$	No
NUCLEAR-ELECTRIC:						
Titan IIIF	$1100\text{-}1200^{d}$	$900\text{-}1000^{d}$	$900\text{-}1000^{d}$	$800\text{-}900^{d}$	$3\frac{1}{2}\text{-}4^{y}$	No
Titan IIIF/Centaur	NA	NA	NA	NA	NA	$8\text{-}9^{y}$

to the ballistic flight mode for Callisto and somewhat longer for the other three satellites.

The nuclear-electric low-thrust flight mode makes possible missions to all six satellites with a Titan-class launch vehicle. Missions to the satellites of Jupiter and Saturn require the Titan IIIF vehicle (seven-segment solids). A mission to Triton requires the Titan IIIF/Centaur. Flight time requirements to the Galilean satellites are somewhat longer than the ballistic and solar-electric counterparts, this being attributed to the use of spiral earth-departure and Jupiter-approach maneuvers employed with the nuclear-electric mode. For a nuclear-electric flight to Titan (Saturn) this time deficit is made up on the interplanetary transfer. For a Triton (Neptune) mission the flight time is from 2 to 4 years shorter with nuclear-electric propulsion.

Based on the results of this study, it is concluded that composite orbiter/lander missions to the outer planet satellites are deserving of further study. Specifically, we recommend a prephase-A mission study for missions to Ganymede (Jupiter) and Titan (Saturn). Primary emphasis should be given to definition of scientific objectives, instruments, subsystem requirements, operations, propulsion system tradeoffs, and comparison of the exploration potential of a soft-lander versus multiple rough landers.

Report No. M-26

"MERCURY ORBITER MISSION STUDY"
An Interim Report
By D. A. Klopp and W. C. Wells
November 1970

The primary purpose of this nearly-completed study is to investigate the utility of solar-electric low-thrust propulsion as applied to an early Mercury orbiter mission. The study is scheduled for completion during December 1970. The major sub-tasks of the study are:

1. Definition of scientific objectives and measurements which might be achieved by an early orbiter mission.

2. Preliminary definition of science payload and spacecraft bus characteristics.

3. Survey of Earth-Mercury low-thrust trajectory optimization.

4. Comparison of solar-electric low-thrust mission capabilities to ballistic mode mission capabilities.

These sub-tasks have been discussed in greater detail in the previous semi-annual report, which summarized the current status of planetological science as applied to Mercury, the scientific objectives which might be achieved during an early orbiter mission, and specific scientific objectives. It has been concluded that visual imaging experiments are likely to be the most useful type of experiments. Consequently, mission analysis is likely to emphasize those mission modes most suitable for performing visual imaging experiments.

Science Payload

A preliminary estimate of a science payload representative of a first-generation orbiter consists of the six instruments listed in Table 2-7. The instrument characteristics given in the table are based upon planet observation from a circular polar orbit of 500 km altitude. The $1\frac{1}{2}$-inch vidicon camera will provide single-frame pictures of a 600 x 600 km surface area at a ground resolution of 1-3 km, depending upon the scene contrast. Vertical height differences on the planet's surface of 500 meters should be detectable at low sun angles. Complete visual coverage of Mercury's surface will require 180 days in orbit.

The infrared line scanner provides thermal mapping of Mercury's surface under both daylight and night conditions. It can be operated simultaneously with the television camera thus providing both visual and thermal imagery (3-50 μm) of the same surface areas. Temperature differences of five deg K can be detected reliably at ground resolutions of three km. The television camera together with the infrared line scanner meet the major imaging requirements for an early orbital mission.

Both the infrared and microwave radiometers acquire data from a three km wide swath directly underneath the orbiting spacecraft and aligned with the orbital ground trace. The ten-channel infrared instrument operates in the spectral region 2 to 30 μm, supplementing data from the wide band infrared scanner, and may be able to detect compositional differences on the surface. The microwave radiometer, operating at 40 GHz, is capable of detecting brightness temperature differences of less than one deg K. The radar altimeter utilizes the same antenna as the microwave radiometer and should be capable of measuring local altitude differences of less than a few hundred meters.

TABLE 2-7

REPRESENTATIVE SCIENCE PAYLOAD

MERCURY ORBITER

INSTRUMENT	WEIGHT (lbs.)	AVERAGE POWER (watts)	DATA ACQUISITION RATE (kbps)
1. TELEVISON CAMERA	24	24	60
2. INFRARED LINE SCANNER	11	7	5
3. TEN-CHANNEL INFRARED RADIOMETER	20	15	0.1
4. MICROWAVE RADIOMETER	37	20	0.01
5. RADAR ALTIMETER	20	10	0.01
6. ULTRAVIOLET SPECTROMETER	8	6	0.1
SCIENCE PAYLOAD	120	70	65

Such data would supplement topographic relief information derived from analysis of the television imagery. The ultraviolet spectrometer is included in the nature of a contingency, but could provide data useful in deducing the composition of Mercury's tenuous atmosphere. The total science package is estimated to weigh about 120 lbs. (55 kg). During normal operation, an average power requirement of about 70 watts is anticipated, and data is acquired at the rate of about 65 kilobits per second. Currently the instrument selection and design is being refined, but the weight, power requirement, and data acquisition rate of the science package finally selected is not expected to differ materially from the estimates given here.

Orbiter Bus

Having established a representative science payload and data rate, the size of the supporting subsystems may be estimated with preliminary results shown in Table 2-8. The rate at which data must be transmitted from the orbiter to Earth is largely dominated by the operational profile of the television camera system. The estimates presented here are based upon a one-orbit data load of 10^8 bits with new data required every 23 orbits. This profile can be accommodated by transmitting to the Goldstone antenna two hours every day at a rate of 12 kilobits per second. These estimates are currently being refined. For example, the time dependence of the Sun-Earth-Mercury angle is being studied to determine its influence upon the opportunities for data transmission. Therefore the 835 lb. (380 kg) orbiter weight shown in the table should be regarded as a preliminary estimate which may be revised, but is useful in focusing upon the range of interest (800 to 1000 lb. or 350 to 450 kg) with regard to the interplanetary transfer.

TABLE 2-8

REPRESENTATIVE MERCURY ORBITER SUBSYSTEMS

SUBSYSTEM	WEIGHT (lbs.)	POWER (watts)
SCIENCE PAYLOAD	120	70
STRUCTURE, PYRO, THERMAL	200	–
CABLING	70	–
RADIO, COMMAND, SEQUENCER	65	40
DATA CONDITIONING	50	30
DATA RECORDER	15	10
ATTITUDE CONTROL	100	20
S-BAND ANTENNA	30	–
TRANSMITTER	20	120
SOLAR PANELS	35	–
BATTERIES	80	–
POWER CONDITIONING	50	–
TOTAL ORBITER	835	290

Ballistic Flight Modes

Earth-Mercury ballistic and swing-by trajectories suitable for orbital missions in the time frame 1980-2000 have been identified by Manning ("Trajectory Modes for Manned and Unmanned Missions to Mercury", Journal of Spacecraft and Rockets, Vol.4, No.9, September 1967, pp.1128-1135). In each launch year, the optimum opportunity is presumed to be that opportunity which minimizes the sum of the Earth departure ΔV and the Mercury capture ΔV. Optimum trajectories generally involve arriving near Mercury's nodes because of the inclination of Mercury's orbit. The direct ballistic annual minimum ΔV opportunities repeat on a 13-yr. cycle. Assuming a 1000 lb. orbiter in a 500 km altitude circular orbit at Mercury, the total spacecraft weight which must be parked in a 100 nm altitude Earth orbit prior to the trans-Mercury injection may be estimated. The results are shown in Figure 2-4 over the 13-yr. cycle from 1980 to 1993. These results are based on a 315 I_{sp} for the Venus swing-by and Mercury capture propulsive stages, and a 445 sec I_{sp} for the Earth departure stage. Both two and three stage chemical retro maneuvers at Mercury were examined. For comparison, the figure also shows, by horizontal dashed lines, the parking orbit delivery capabilities of selected launch vehicles. The 1980, 1982, 1983, 1988, and 1989 launch opportunity results are based on a Venus swing-by maneuver; the other opportunities utilize the direct ballistic mode. The most favorable opportunity is the 1988 powered Venus swing-by, although the 1000 lb. Mercury payload is slightly beyond the capability of a Titan IIID(7)/Centaur.

Solar-Electric Flight Modes

Low-thrust interplanetary mission modes are often characterized by the performance index J, defined as the square of the thrust acceleration vector integrated over the time of

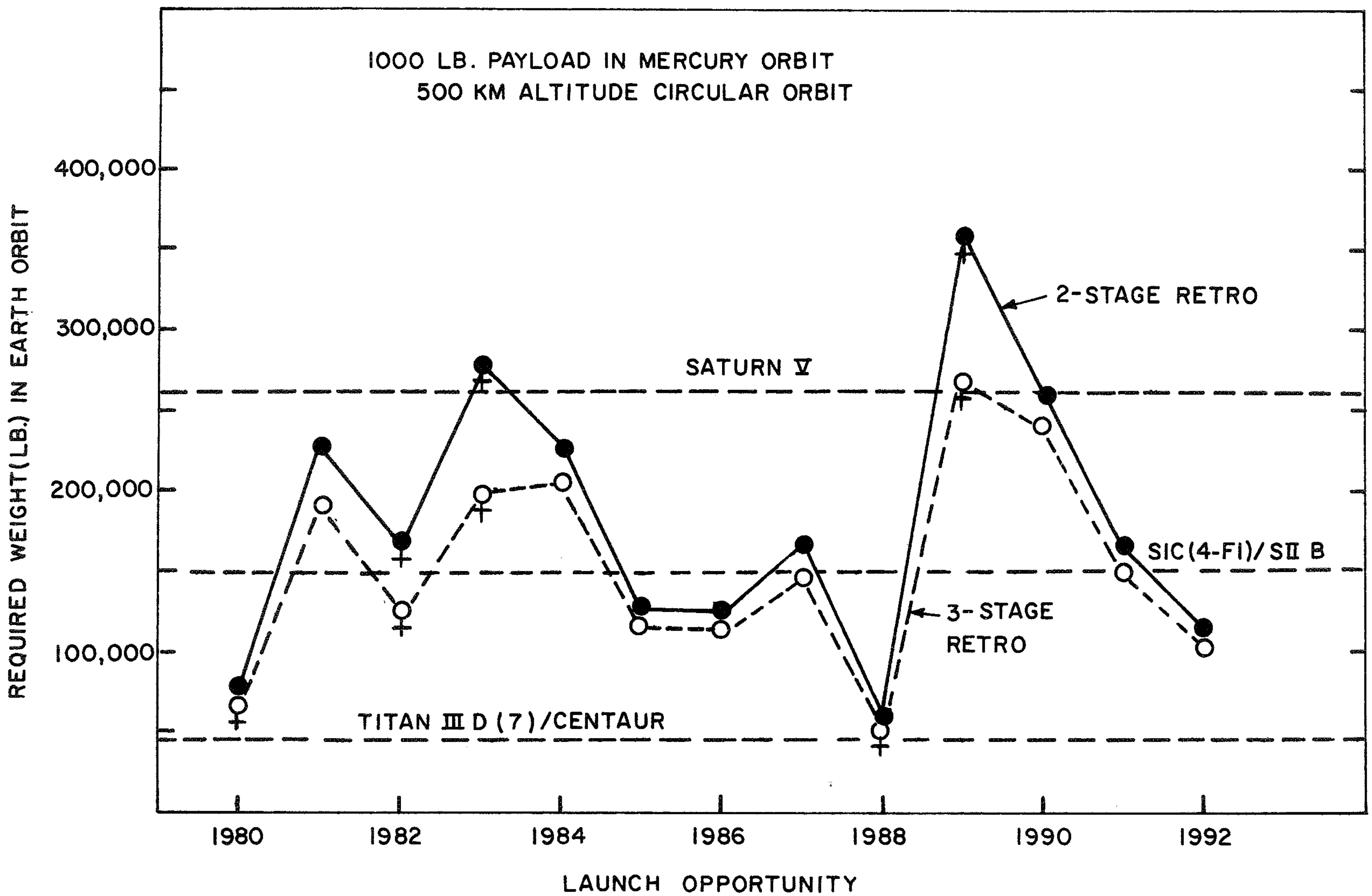


FIGURE 2-4. MERCURY ORBITAL MISSIONS(CIRCULAR ORBIT)

flight. Payload capability goes inversely with J, that is, for a specified launch vehicle, the larger the value of J the smaller the delivered payload. Thus J is analogous, in some respects, to the ideal velocity or the value of C_3 with reference to a ballistic mission. For fixed values of the hyperbolic launch excess velocity (VHL), the hyperbolic arrival excess velocity (VHP), and the time of interplanetary flight, launch opportunities may be recognized by examining the dependence of J upon the date of launch. Since the time of flight is fixed, J may be equally well plotted against the date of arrival, rather than the date of launch, as in Figure 2-5. For each arrival date, the figure shows the minimum value of J which can be achieved by an optimum thrust vector control program in which both the magnitude and the direction of the thrust vector can be varied at will. For this reason, the J shown on the ordinate is J_v (m^2/sec^3), the variable thrust J. The results shown assume a 400 day trip time, a VHL of five km/sec, and a VHP of zero km/sec (rendezvous at Mercury). These values are near optimum for a Titan IIID (5)/Centaur launch vehicle. In general, it has been found that there are about three relatively good launch opportunities in each calendar year. Unlike ballistic trajectories, the optimum arrival date appears to be completely unrelated to Mercury's line of nodes or its apse line. For this reason, the results shown are expected to be generally typical of any launch year. It should be emphasized that in obtaining these results, Mercury's orbit was assumed to be both eccentric and inclined to the ecliptic plane. That is, although earlier studies had also indicated that the optimum arrival date is not correlated with the apse line or line of nodes, it could be argued that such correlation was obscured by the assumption that Mercury's orbit is circular and coplanar with the ecliptic. This is not the case with the results shown here.

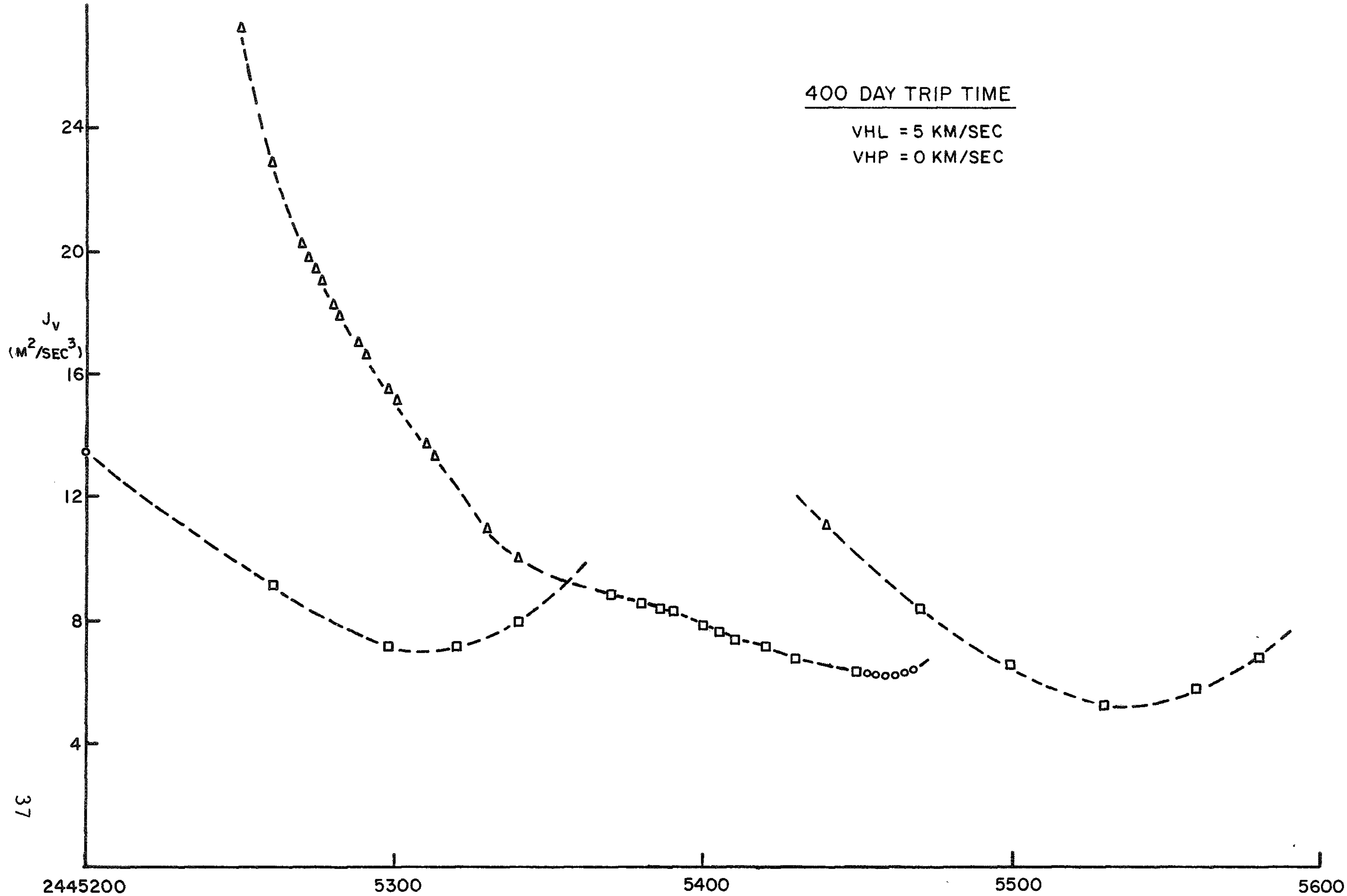


FIGURE 2-5. J_V VS. ARRIVAL DATE

Another property of the solar-electric low-thrust launched opportunities indicated in the figure is the relative broadness of the low-thrust launched opportunities, as compared to typical ballistic launch opportunities. For small departures from the optimum (minimum) value of J, the corresponding decrease in delivered payload is proportional to the increase in J.

The size of the delivered payload depends upon many factors not indicated in Figure 2-5, such as launch vehicle and solar-electric propulsion stage characteristics. Results obtained thus far has been based upon the use of the Titan 3X (1205)/Centaur launch vehicle and optimizing the VHL, the VHP, the arrival (or launch) date, and the solar-electric stage power and specific impulse to deliver the maximum orbit payload. The payload results are shown in Figure2-6 as a function of trip time from 300 to 400 days. Longer trip times would result in even larger payloads, but were not investigated. It should also be noted that jettisoning the entire solar electric propulsion stage prior to the orbit capture maneuver increases the injected payload by about 1000 lbs. (450 kg). The injected payload also depends upon the orbit chosen at Mercury. Figure 2-7 shows injected payloads (assuming jettisoning of the solar-electric stage) as a function of periapse and apoapse radius for a 400 day trip time. Finally, Figure 2-8 shows the optimum power (at one AU) and specific impulse of the solar-electric stage as a function of trip time.

Remaining Tasks

As mentioned above, the science payload and spacecraft subsystem sizing are currently being reviewed. In addition, the solar-electric mode will be investigated for off-optimum (that is, smaller) power levels and for other launch vehicles.

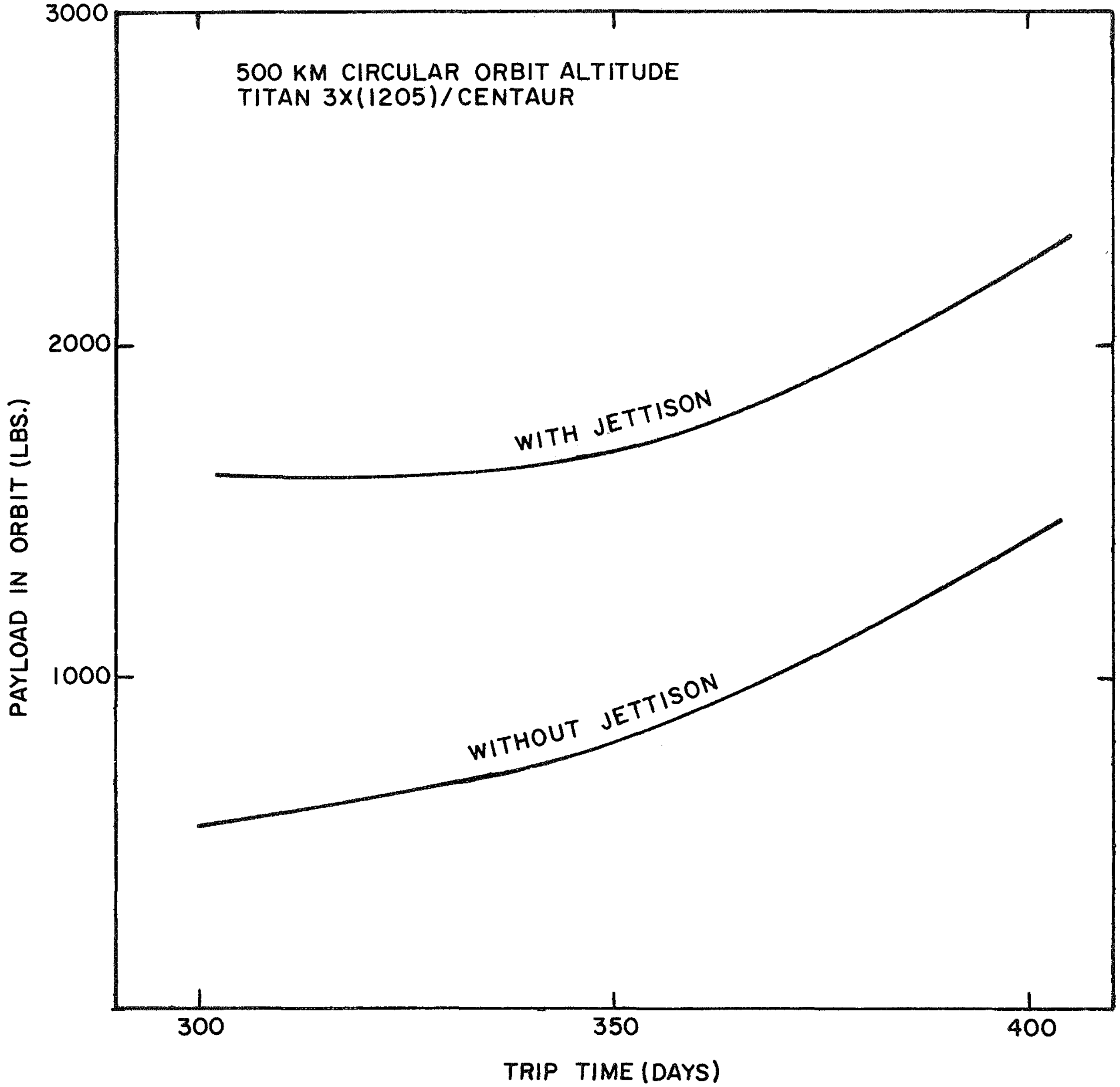


FIGURE 2-6. MERCURY ORBITER PAYLOAD VS. TRIP TIME.

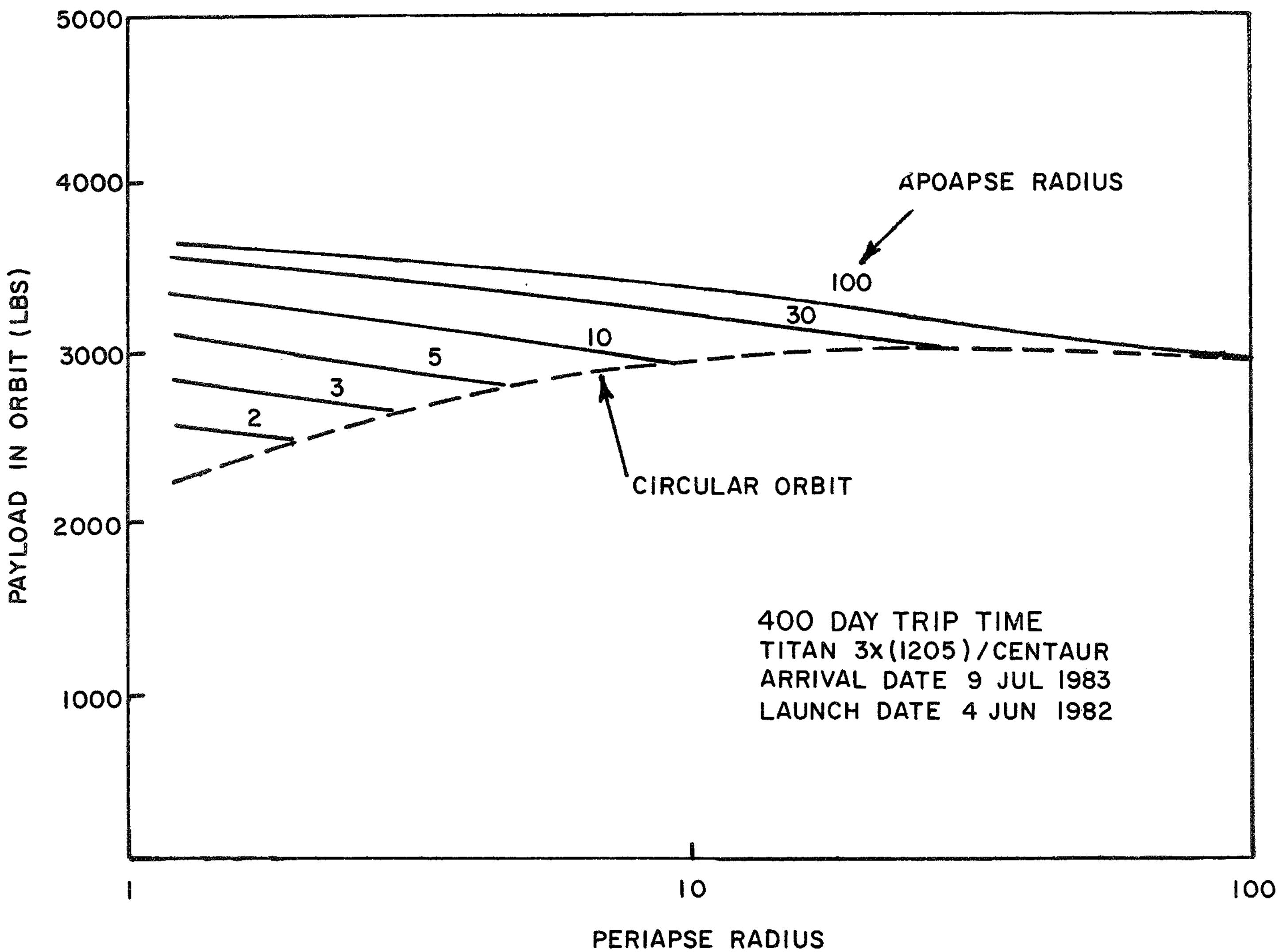


FIGURE 2-7. INJECTED PAYLOAD IN MERCURY ORBIT VS. PERIAPSE AND APOAPSE RADIOS.

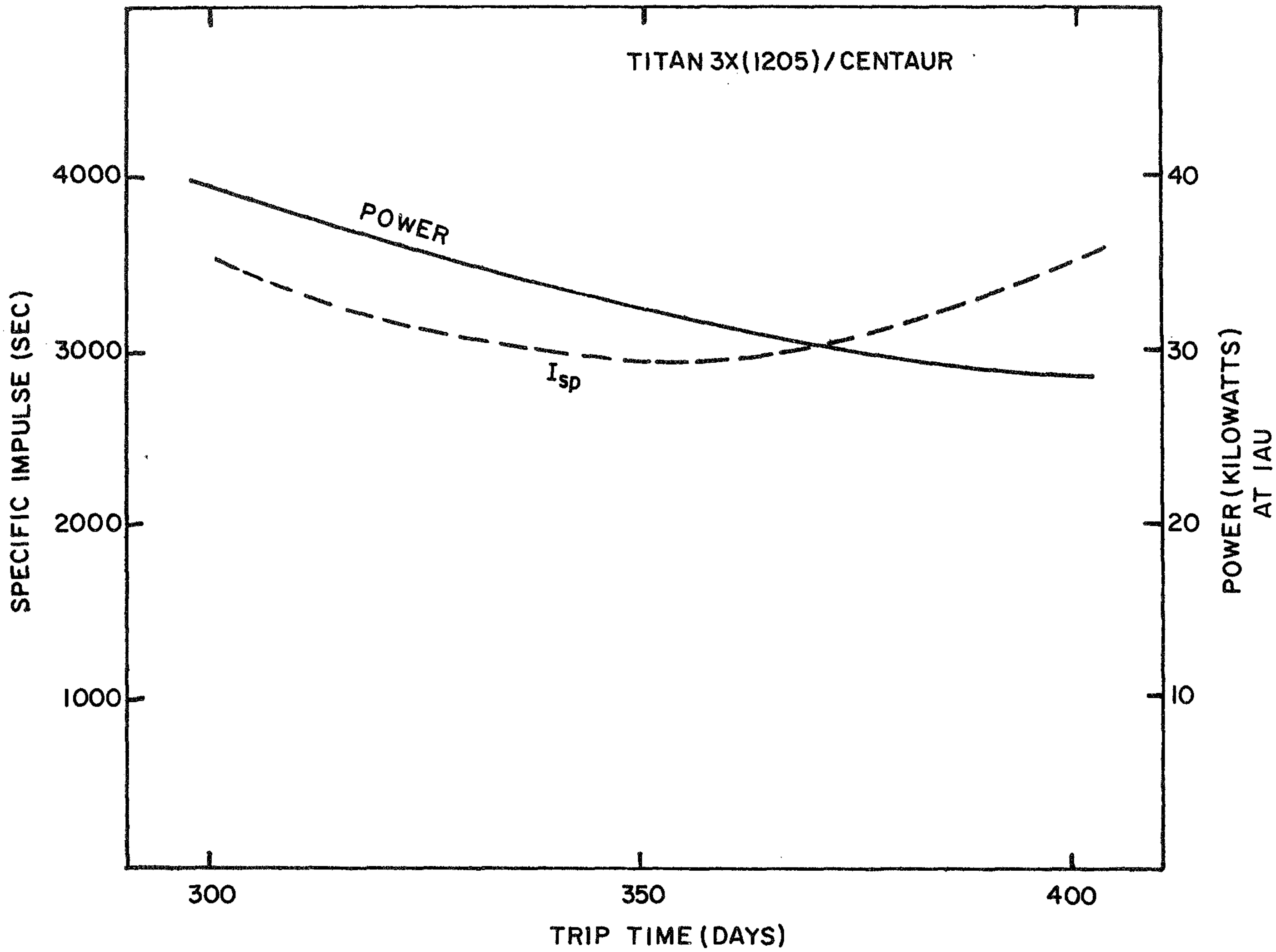


FIGURE 2-8. OPTIMUM POWER AND SPECIFIC IMPULSE OF THE SOLAR-ELECTRIC STAGE VS. TRIP TIME.

An oral presentation summarizing the main results of the study is scheduled for December 16, while a preliminary draft of the study final report is scheduled for completion shortly after the first of the year.

Report No. M-27
"ATMOSPHERIC ENTRY AT URANUS AND NEPTUNE"
An Interim Report by J. I. Waters and M. J. Price
November 1970.

The objective of this study is that of defining survivable probes for entry into the atmospheres of Uranus and Neptune. The JPL-TOPS is used as a guideline design of the parent spacecraft. The study is restricted to two mission opportunities:

the 1979 J-U-N Grand Tour and
the 1981 S-U-N Grand Tour.

The first part of the study deals with the choice of planetary atmospheric models. These models are used in the study's second half, dealing with entry probe trajectories. Finally a tentative probe design will be suggested.

Models of the physical structures of the atmospheres of Uranus and Neptune are based primarily on recent work by Belton, McElroy, and Price (to be published). From an analysis of available spectroscopic, photometric and radio observational data, these two planets were shown to be very similar, being surrounded by deep, essentially pure, H_2 atmospheres. Helium is a minor constituent, and both atmospheres are clear of cloud particles at least down to the effective level of penetration of solar photons. Radii have been obtained for the atmospheres from albedo studies and scale heights derived on the assumption that the atmospheres are in radiative equilibrium with the solar flux. Preliminary selection of the science objectives for initial probe missions has been made and a candidate science instrument payload assembled. The instrument payload, listed in Table 2-9, weighs 22 pounds not including a power supply and communications equipment.

TABLE 2-9

CANDIDATE INSTRUMENTS FOR SURVIVABLE PROBE MISSIONS TO URANUS AND NEPTUNE

MEASURABLE	INSTRUMENT	WEIGHT (LBS.)
CHEMICAL COMPOSITION	RAM SPECTROMETER	3.3
	NEUTRAL MASS SPECTROMETER	6.0
	ION MASS SPECTROMETER	3.3
DENSITY	LOW-g ACCELEROMETER	1.8
	HIGH-g ACCELEROMETER	1.8
PRESSURE	RAM PRESSURE GAUGE	2.9
	BAROGRAPH	2.2
TEMPERATURE	THERMOMETER	0.7
	TOTAL	22.0

The trajectory study began with the choice of the 1979 J-U-N and 1981 S-U-N Grand Tour mission opportunities as typical interplanetary trajectories. Aerodynamic velocity and flight path angle at entry have been studied as functions of deflection time along the approach paths. Entry conditions resulting from deflection increments of less than 100 km/sec have been found which include entry angles between -20° and -40°. These entry conditions have been forwarded to the Ames Research Center (NASA) along with selected vehicle parameters. Heat shield ablation and insulation mass will be found for the selected conditions by ARC.

A preliminary look at spacecraft-probe communication angles and times suggests that this may be a critical problem, particularly at Uranus. This area will be examined in detail as part of a study of post-entry descent trajectories which is now under way. Having obtained heat shield, pressure vessel and parachute weight requirements a preliminary probe design will be suggested which will attempt to meet communication requirements and science objectives within total probe weight constraints.

Technical Memorandum No. M-28

"INTERSTELLAR MISSIONS"

An Interim Report by R. Brandenburg

October 1970.

Interstellar exploration is the logical extension of solar system exploration, and it is to this end that advanced planning must ultimately be directed. This short memo is designed to be a preliminary, though comprehensive, look at the various aspects of interstellar travel. By tying together much of the previous relevant work in this field, this study will hopefully serve as a base from which more specific and ambitious work on interstellar missions can profit. The study is divided into three major tasks:

- Examine the solar "neighborhood" for interesting or "promising" stellar systems to serve as targets for unmanned interstellar probes.
- Assess the applicability of current and proposed advanced space propulsion systems for interstellar missions.
- Determine the gross requirements for missions to the selected stellar systems using the most favorable propulsion systems (i.e., provide a wide survey giving an indication of the mass ratios, flight times, accelerations, and navigational problems involved.)

The first task has been completed. This was essentially the selection of a few target stars at which to aim our interstellar probes. To do this a sphere with a radius of 20 light-years [1] with the sun at its center was chosen to be the "solar neighborhood". Within this volume of space ($\sim 2.85 \times 10^{43}$ km^3) there are sixty observable [2] stellar systems, including the sun. These systems are listed in Table 2-10. From these sixty systems twelve were chosen as targets for interstellar probes based on three criteria:

1) Similarity to the sun. The primary purpose of interstellar probes should be the search for planets, preferably planets capable of sustaining life [3]. Based on the one data point available (the existence of the solar system) stars of approximately the sun's radius and temperature may have planetary systems. For this reason all F, G, and K type stars were considered as possible targets.

2) Existence of unseen companions. A star whose motion is perturbed by a large dark companion may possibly have smaller planets orbiting it or the dark companion. Barnard's Star is the classical

---

1 1 light - year ≅ 63,000 AU or ≈ 9.4×10^{12} km.

2 There may be more than sixty systems if objects of large mass, but extremely low luminosity are counted, but since these objects are currently undetectable they cannot be included in our survey.

3 First generation interstellar probes should be designed to search for planets capable of sustaining carbon-based life within a specific temperature zone ($273° \pm X°$ K) around a parent star (dependent on the luminosity and radius of the parent star). Life based on another element, such as silicon, or existing upon a large, massive, warm body distant from any star may certainly be possible. However the tremendous difficulties involved in searching for conditions amicable to known forms of life are small compared to those involved in searching for unknown conditions favorable to unknown forms of life.

TABLE 2-10

STELLAR SYSTEMS WITHIN 20 LIGHT-YEARS OF THE SUN

STELLAR SYSTEM	DISTANCE (light-years)	RIGHT ASCENSION	DECLINATION	SPECTRAL TYPE	MASS ($M_\odot$)	RADIUS ($M_\odot$)	LUMINOSITY (SUN=1)
α Centauri	4.3	$14^h36^m.2$	-60°38'	G0(K5,M5)	1.09(0.88,0.1)	1.23(0.87,-)	1.0(0.28,0.000052)
Barnard's Star	6.0	$17^h55^m.4$	+ 4°33'	M5	0.15	~0.12	0.00040
Wolf 359	7.7	$10^h54^m.2$	+ 7°20'	M6	~0.20	~0.03	0.000017
Luyten 726-8	7.9	$1^h36^m.4$	-18°13'	M6 (M6)	—	~0.05(~0.04)	0.00004(0.00003)
Lalande 21185	8.2	$11^h\ 0^m.6$	+36°18'	M2	0.35	~0.35	0.0048?
Sirius	8.7	$6^h42^m.9$	-16°39'	A0(wd)	2.31(0.98)	1.8(0.022)	23.0(0.008)
Ross 154	9.3	$18^h46^m.7$	-23°53'	M5	~0.31	~0.12	0.00036
Ross 248	10.3	$23^h39^m.4$	+43°55'	M6	~0.25	~0.07	0.00010
ε Eridani	10.8	$3^h30^m.6$	- 9°38'	K2	0.80	0.90	0.25
Ross 128	10.9	$11^h45^m.1$	+ 1° 7'	M5	~0.31	~0.10	0.00030
61 Cygni	11.1	$21^h\ 4^m.7$	+38°30'	K6 (M0)	0.59(0.50)	0.70(0.80)	0.052(0.028)
Luyten 789-6	11.2	$22^h35^m.7$	-15°37'	M6	~0.25	~0.08	0.00012
Procyon	11.3	$7^h36^m.7$	+ 5°21'	F5 (wd)	1.75(0.64)	1.7(0.01)	5.8(0.00044)
ε Indi	11.4	$21^h59^m.6$	-57° 0'	K5	0.71	1.0	0.12
Σ 2398	11.6	$18^h42^m.2$	+59°33'	M4 (M4)	—	~0.28(~0.20)	0.0028(0.0013)
Groombridge 34	11.7	$0^h15^m.5$	+43°44'	M2 (M4)	0.38	~0.38(~0.11)	0.0058(0.00044)
τ Ceti	11.8	$1^h41^m.7$	-16°12'	G4	0.82	~0.67	0.36
Lacaille 9352	11.9	$23^h\ 2^m.6$	-36° 9'	M2	0.47	~0.57	0.013
BD + 5°1668	12.4	$7^h24^m.7$	+ 5°29'	M4	~0.38	~0.16	0.0010
Lacaille 8760	12.8	$21^h14^m.3$	-39° 4'	M1	0.54	~0.82	0.028
Kapteyn's Star	13.0	$5^h\ 9^m.7$	-45° 0'	M0	~0.44	~0.24	0.0025
Kruger 60	13.1	$22^h26^m.3$	+57°27'	M4 (M5e)	0.27(0.16)	0.51 (-)	0.0013(0.00033)
Ross 614	13.1	$6^h26^m.8$	- 2°47'	M5e	0.14	~0.14	0.00052
BD - 12°4523	13.4	$16^h27^m.5$	-12°32'	M5	~0.38	~0.22	0.0013
van Maanen's Star	13.8	$0^h46^m.5$	+ 5°10'	wdF	-	-	0.00016
Wolf 424	14.6	$12^h30^m.9$	+ 9°18'	M6 (M6)	-	~0.09(~0.09)	0.00014(0.00014)
Groombridge 1618	14.7	$10^h\ 8^m.3$	+ 49°42'	K5	0.56	~0.5	0.030
CD - 37°15492	14.9	$0^h\ 2^m.5$	-37°36'	M3	0.39	~0.4	0.0058
CD - 46°11540	15.3	$17^h24^m.9$	-46°51'	M4	~0.44	~0.25	0.0023
BD + 20°2465	15.4	$10^h16^m.9$	+20° 7'	M4	~0.44	~0.28	0.0028
CD - 44°11909	15.6	$17^h33^m.5$	-44°16'	M5	~0.34	~0.15	0.00058
CD - 49°13515	15.6	$21^h30^m.2$	-49°13'	M3	0.37	~0.34	0.0044
AOe 17415-6	15.8	$17^h36^m.7$	+68°23'	M3	0.35	~0.39	0.0040
Ross 780	15.8	$22^h50^m.5$	-14°31'	M5	~0.39	~0.23	0.0014
Lalande 25372	15.9	$13^h43^m.2$	+15°10'	M2	-	~0.40	0.0063
CC 658	16.0	$11^h42^m.7$	-64°33'	wd	-	-	0.0008
o Eridani	16.3	$4^h13^m.0$	- 7°44'	K0 (wdA,M5)	0.11(0.44,0.21)	0.7(0.018,0.43)	0.30(0.0040,0.0008)
70 Ophiuchi	16.4	$18^h\ 2^m.9$	+ 2°31'	K1 (K5)	0.89(0.68)	~1.03(~0.84)	0.40(0.083)
Altair	16.5	$19^h48^m.3$	+ 8°44'	A5	590.0	1.2	8.3
BD + 43°4305	16.5	$22^h44^m.7$	+44° 5'	M5	0.26	~0.24	0.0016
AC 79°3888	16.6	$11^h44^m.3$	+78°57'	M4	~0.35	~0.15	0.0008
+15°2620	16.9	$13^h41^m.0$	+15°26'	M1	0.42	~0.50	~0.01
η Cassiopeiae	18.0	$0^h43^m.0$	+57°17'	F9 (K6)	0.85(0.52)	0.84(0.07)	1.0(~0.03)
σ Draconis	18.2	$19^h33^m.0$	+69°29'	G9	0.82	~0.28	~0.4
36 Ophiuchi	18.2	$17^h\ 9^m.0$	-26°27'	K2 (K1, K6)	0.77(0.76,0.63)	~0.90(~0.82,~0.90)	~0.26(~0.26,0.09)
HR 7703	18.6	$20^h\ 5^m.0$	-36°21'	K2 (M5)	0.76(~0.35)	~0.80(~0.14)	~0.20(~0.0008)
HR 5568	18.7	$14^h51^m.6$	-20°58'	K4 (M0)	0.70(0.50)	~0.87(~0.61)	~0.14(~0.017)
Lalande 21258	19.2	$11^h\ 0^m.5$	+44° 2'	M0 (M7)	0.43	~0.47(~0.05)	~0.01(~0.00004)
-21°1377	19.2	$6^h\ 6^m.0$	-21°49'	M0	0.455	~0.59	~0.016
Luyten 97-12	19.2	$7^h53^m.0$	-67°30'	wd	-	-	~0.00003
δ Pavonis	19.2	$19^h59^m.0$	-66°26'	G7	0.98	~1.07	~1.0
Luyten 347-14	19.3	$19^h13^m.0$	-45°42'	M7	0.26	~0.08	~0.0001
+4°4048	19.4	$19^h12^m.0$	+ 5° 2'	M3 (M5)	0.39	~0.43(~0.008)	~0.007(~0.000002)
I (UC 48)	19.5	$17^h38^m.0$	-57°14'	M	0.14	-	~0.0002
-40°9712	19.5	$15^h26^m.0$	-40°54'	M4	0.44	~0.29	~0.003
Ross 47	19.9	$5^h36^m.0$	+12°29'	M5	0.35	~0.17	~0.0008
Luyten 745-46	19.9	$7^h36^m.0$	-17°10'	wdF (M)	-	-	~0.0002(~0.000003)
HD 36395	20.0	$5^h26^m.0$	- 3°42'	M1	0.51	~0.69	~0.02
Wolf 294	20.0	$6^h48^m.0$	+33°24'	M3	0.49	~0.46	~0.008

example of this type of system, having one or two unseen companions about the same mass as Jupiter. Any stellar system thought to have a low-mass dark companion was considered as a possible target.

3) Instability of planetary orbits within a star's "life-zone" due to perturbation by other components of multiple star systems. If a range of temperatures is assumed for the existence of life the extent of a zone about a star can be computed (using the star's radius and luminosity) within which the interstellar probe should search for planets. If in multiple star systems, perturbations cause planetary orbits within these zones to be unstable, the system was not considered to be a good target for the purposes of this study.

The twelve selected stellar targets are listed in Table 2-11, and their positions, as viewed from the earth, plotted on a sky map in Figure 2-9.

The second task, the examination of propulsion systems has just begun.

In order to get some indication of the propulsion system requirements for interstellar probes, flight times to several of the selected target stellar systems at several velocities are shown in Figure 2-10 (for this figure acceleration and deceleration times ranged from a few months, for the low velocities, to several years, for the higher velocities). Note that the time scale on the ordinate exceeds the length of recorded history. If an upper limit to mission duration is set at several human lifetimes it is clear from this figure that velocities in at least the lower relativistic region ($v > 0.1c$) must be obtainable. Since we are dealing with unmanned probes the relativistic time dilation or slow-down experienced aboard

TABLE 2-II

TARGET STELLAR SYSTEMS

STELLAR SYSTEM	DISTANCE (LIGHT-YEARS)	REMARKS
α CENTAURI	4.3	Three component system; α Cen A almost identical to the Sun, B (near A, r_p = 11.2 AU, r_a = 35.6 AU) smaller and cooler, C a red dwarf quite distant from A & B.
BARNARD'S STAR	6.0	Small cool red dwarf. May have one ($m \cong 1.7\ m_{♃}^*$, $a \cong 4.5$ AU) or two ($m_1 \cong 1.1\ m_{♃}$, $a \cong 4.7$ AU; $m_2 \cong 0.8\ m_{♃}$ $a \cong 2.8$ AU) unseen companions.
ϵ ERIDANI	10.8	Single star, cooler and smaller than the Sun.
61 CYGNI	11.1	A double system which may have a third unseen component ($m \cong 16 m_{♃}$).
PROCYON	11.3	Double system with a hot, yellow-white main component, and a small, faint (perhaps white dwarf) companion.
ϵ INDI	11.4	Single star, cooler than the Sun.
τ CETI	11.8	Single star, similar to the Sun.
GROOMBRIDGE 1618	14.7	Single star, cooler than and about half the size of the Sun.
70 OPHIUCHI	16.4	Double system, A & B revolve about each other with a period of ~ 88 yr. ($e \cong 0.5$, $a = 22.8$ AU. May be a third dark companion.
η CASSIOPEIAE	18.0	Double system. Component A is nearly identical to the Sun. A & B revolve about each other with a period of ~ 500 yr. ($e \cong 0.53$, $a \cong 70$ AU). May be a third unseen companion.
σ DRACONIS	18.2	Single star, slightly cooler than the Sun.
δ PAVONIS	19.2	Single star, similar to the Sun.

* $m_{♃}$ = MASS OF JUPITER, $\approx 1.9 \times 10^{27}$ kg

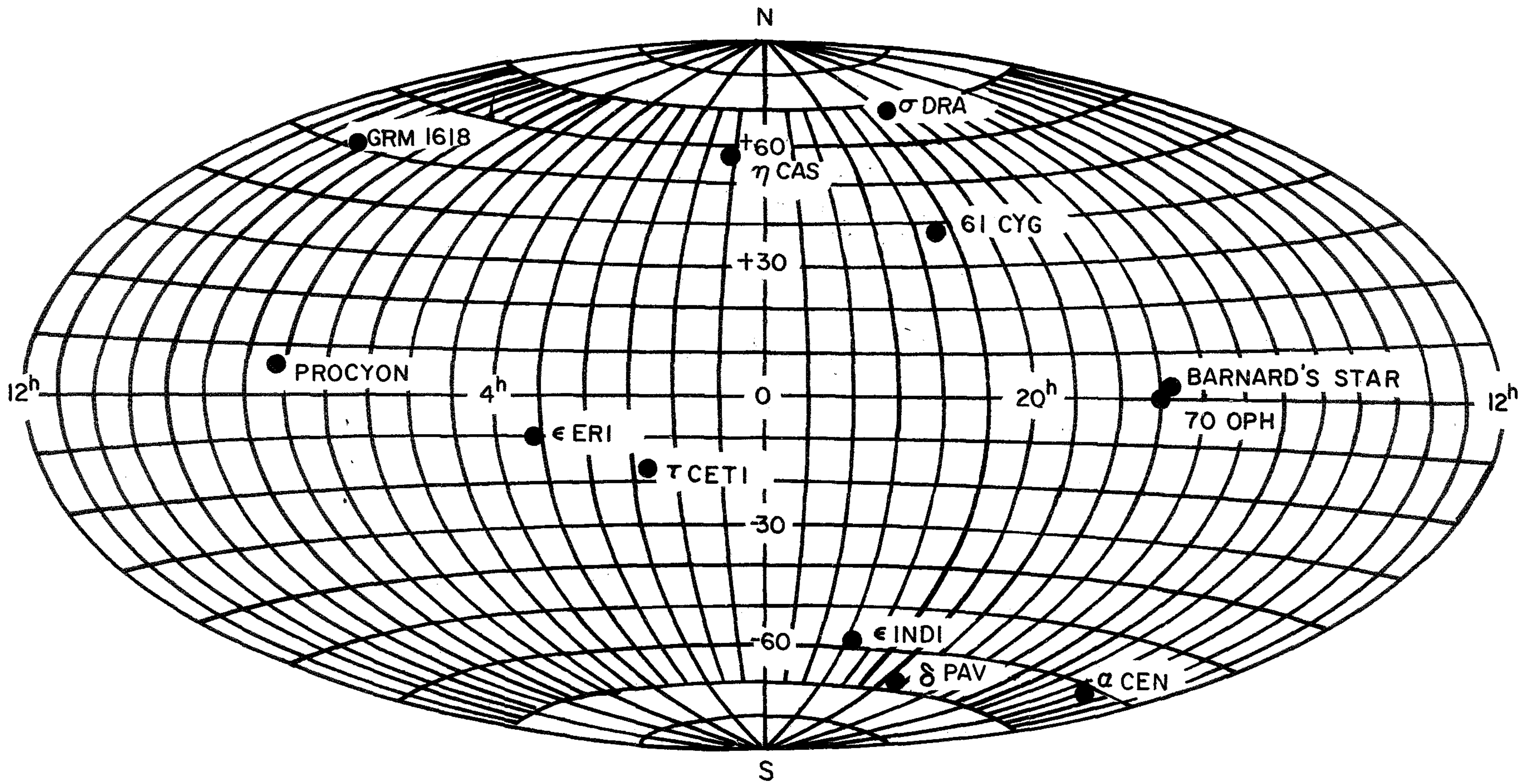


FIGURE 2-9. TARGET STELLAR SYSTEMS

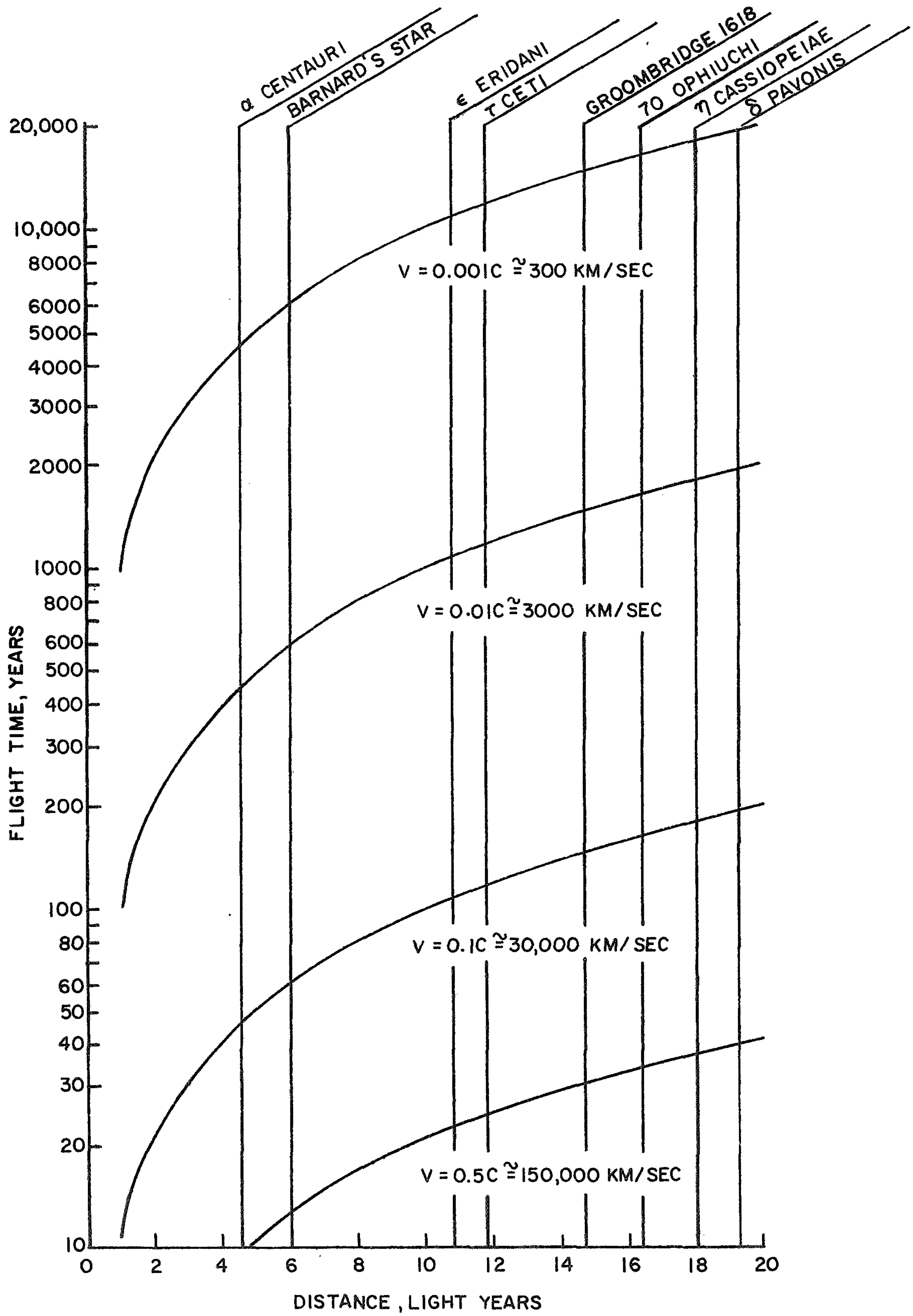


FIGURE 2-10. FLIGHT TIMES TO SEVERAL TARGET STELLAR SYSTEMS

the probe is useful only in reducing the reliability requirements of the spacecraft, and not in reducing the apparent flight time for an astronaut.

The propulsion systems to be assessed in this study are:

- Nuclear electric propulsion
- Nuclear Pulse (Bomb) propulsion [4]
- Nuclear (Fission and Fusion) staged rocket
- Fusion rocket/X-ray pumped gas laser drive[5]
- Fusion rocket/X-ray powered ion drive[5]
- Bussard Interstellar Ramjet[6]
- Matter - anti matter photon propulsion[7]

---

[4] The nuclear pulse rocket, theoretically developed in the early 1960's at Gulf-General Atomic (Project Orion), employs a large number of small thermonuclear bombs, ejected out the rear of the spacecraft and exploded every few seconds. A large ablation shield/pusher-plate absorbs the momentum and with the aid of a large shock absorber transfers a constant acceleration to the payload end of the spacecraft.

[5] A way to utilize the energy of X-rays produced in a fusion rocket engine is presented in "Some Aspects of Thermonuclear Propulsion" by G. L. Matloff and H. H. Chiu in The Journal of the Astronautical Sciences, Vol. XVIII, No.1, pp.57-62, July - August 1970.

[6] The interstellar ramjet magnetically funnels interstellar hydrogen into its fusion reactor, greatly reducing the amount of transported terrestrial fuel.

[7] In this system matter and anti-matter are brought together, allowed to annihilate one another (baryon conservation), and the resultant radiant energy reflected and collimated into a unidirectional thrust beam. The problem for this system is the production and containment of sizeable quantities of anti-matter.

The nuclear fission or fusion staged rocket appears at this time to be the only technologically feasible propulsion system possible within the next several decades which can attain the high accelerations and relativistic velocities necessary in interstellar travel. Using a one g acceleration (and deceleration at target) a five stage uranium fission rocket may be capable of achieving velocities of 0.3 c with a mass ratio (initial rest mass/final rest mass at burnout) of about 10^6 to 10^7. The flight time [8] to α Centauri would then be about 15-16 years, and to δ Pavonis, ~ 65 years. The five stage fusion rocket, also with a mass ratio of 10^6 to 10^7, may be capable of velocities as high as 0.6 c and flight times of ~ 8 years and ~ 32 years to α Centauri and δ Pavonis, respectively.

The third task, the development of an overall view of interstellar missions, should focus on some of the more interesting aspects involved, such as the problems or phenomena associated with navigation and velocity vector control. Navigation at relativistic velocities will be a particular problem. In addition to the shift in relative stellar positions on the celestial sphere due to the probes motion out of the solar system, the doppler effect will cause the spectrum of the stars in the forward direction to shift toward the UV, and those in the backward direction toward the IR. The aberation of light at relativistic velocities will also cause a distortion in the apparent star field; at 0.9 c approximately 90% of the visible stars will appear to the probe to be in the forward hemisphere. All these effects point to the need for a highly sophisticated navigational scheme.

---

8 The total mission time ≥ flight time + distance to the target stellar system in light-years. The second term here allows for data transmission back to earth (4.3 years for α Centauri, 19.2 years for δ Pavonis).

R. L. Forward [9] pointed out the interesting possibility of using the galactic magnetic field to control the interstellar probes' velocity vector. Using a long charged cable a probe could perform extensive midcourse maneuvers, perhaps even circling back into the solar system after its interstellar voyage. The properties of the local galactic magnetic field would have to be more thoroughly known than at present, however, for such a scheme to be workable.

---

9 "Zero Thrust Velocity Vector Control for Interstellar Probes: Lorentz Force Navigation and Circling" R. L. Forward, AIAA Journal, Vol.2, No.5, May 1964.

Report No. M-29

"COMET RENDEZVOUS MISSION STUDY"

An Interim Report by A. L. Friedlander, D. J. Spadoni, and W. C. Wells

November 1970.

The goal of this study is to establish the value and characteristics of a comet rendezvous mission. Previous comet mission studies have considered fly through which typically allows one day within the cometary region. The rendezvous mode permits several hundred days of observations. With an arrival before perihelion the temporal aspect of comet activity can be studied. During that time extensive spatial investigations can also be made.

The study task objectives are listed in Table 2-12. At this time, the review of comet theories and observations will be discussed briefly. Following that will be a summary of low-thrust solar-electric trajectories for the 1980 apparition of P/Encke. Guidance and stationkeeping for the Encke mission will also be discussed. Three other comets recommended in the Comet Rendezvous Opportunities study will also be investigated.

COMET THEORIES AND OBSERVATIONS

Both the justification of comet missions and important design parameters for these missions depend on understanding the theories of comet origin and activity. The four comets in this study are in periodic orbits and it is thought that they have been captured by planetary perturbations. Previous to capture they would have been a new or parabolic comet which had spent most of the last 4.5×10^9 years in a vast cloud of approximately 10^{11} comets located between 20,000 and 100,000 AU from the sun. Only in that way can cometary material survive, for the

TABLE 2-12

COMET RENDEZVOUS MISSIONS

STUDY OBJECTIVE

- o ESTABLISH THE VALUE AND CHARACTERISTICS OF THE RENDEZVOUS MISSION MODE OF COMET STUDY

TASK OBJECTIVES

1. REVIEW THEORIES AND OBSERVATIONS OF COMETS.

2. SELECT AND DESCRIBE INSTRUMENTS THAT WILL PROVIDE A SIGNIFICANT RETURN OF DATA.

3a. DETERMINE TRAJECTORY CHARACTERISTICS FOR 3-IMPULSE BALLISTIC AND GRAVITY-ASSISTED MISSIONS.

b. INVESTIGATE LOW THRUST (SOLAR OR NUCLEAR ELECTRIC) TRAJECTORY CHARACTERISTICS.

4. DETERMINE THE GUIDANCE MANEUVERS REQUIRED BY COMET ORBIT UNCERTAINTIES.

5. SPECIFY TYPICAL STATIONKEEPING OPERATIONS OF THE SPACECRAFT.

6. ESTIMATE SPACECRAFT PARAMETERS (WEIGHT, POWER, DATA RATE) FROM A SUBSYSTEM SCALING ANALYSIS.

7. SYNTHESIZE GOOD COMBINATIONS OF TRAJECTORY CAPABILITY, SCIENTIFIC VALUE AND OPERATIONAL RELIABILITY.

lifetime of a periodic comet is very short compared with the age of the solar system.

The composition of comets is thought to be primative material similar to the early planetary nebula. It would take the form of a small ice nucleus binding together dust and gases which are released when the comet is heated at small solar distances. Some molecules in the cloud or coma about the nucleus can be identified by the resonance fluorescence radiation they emit. Also observed in comet spectra are ions and a solar continuum reflected by dust. Both the ions and the dust are accelerated by sunlight and stream away from the coma to form Type I and Type II tails respectively.

In Table 2-13 the orbital elements of these four comets are listed along with other observed properties. P/Encke is especially interesting because of its short period, and its activity which is a result of its small perihelion distance. Since the 1980 apparition will be the 52nd observed return, there are extensive records on positions, brightness, the spectrum, etc. This comet is frequently cited for its non-gravitational accelerations and its secular decrease in brightness.

P/Halley is by far the most spectacular periodic comet. The long period (76 years) and retrograde orbit are unusual for periodic comets. The dust content is high, and therefore it is hard to identify molecular spectral lines in the presence of the strong continuum. The tail consists of both ion and dust types. It is suspected that the η Acquarids and the Orionids meteorite showers are debris from P/Halley and that the Tourids are due to P/Encke.

TABLE 2-13

OBSERVED COMET PROPERTIES

	P/ENCKE	P/d'ARREST	P/KOPFF	P/HALLEY
ORBITAL ELEMENTS				
Perihelion Date	12/6/80	9/18/82	6/14/83	2/5/86
Period, Yr.	3.303	6.394	6.444	75.993
Eccentricity	0.847	0.622	0.545	0.967
Perihelion Dist., AU	0.339	1.300	1.576	0.587
Inclination	11.95°	19.59°	4.73°	162.24°
Long. of Node	185.98°	138.89°	120.37°	111.86°
Arg. of Perihelion	334.19°	176.93°	162.78°	58.15°
DIMENSIONS				
Radius of Nucleus, km	1.3	1.3	3.2	16.
Radius of Coma, 10^6 km	0.1	0.1	0.05	0.2
Length of Tail, 10^6 km	5.0	>0.1	0.2	30.
Mass g	10^{15}	10^{15}	10^{16}	10^{18}
TYPICAL ACTIVITY				
Brightness (Mo)	11.5	9.5	7.0	4
Coma: begin*	-80	0	-40	-125
Coma: end	+50	+80	+80	+125
Tail: begin	-30	0	-40(?)	-75
Tail: end	-15(?)	+80(?)	+80	+75
Maximum Activity	-15	+30	0	+30(?)
SPECTRUM				
Molecules: C_2	S	M	M	S
C_3	M	M	M	M
CN	S	S	S	S
CH	W	W	W	M
OH	W	W	-	-
NH	W	-	-	-
Ions: CO^+	-	-	-	S
Continuum:	W	W	W	S

(S = strong, M - moderate, W - weak)

* Days From Perihelion

There are eight major questions on the nature of comets which the rendezvous mission should attempt to provide answers for. These questions, listed in Table 2-14 do not consider the origin and previous history of the comet, but it is hoped that by defining the present state, and the processes modifying that state, the past can be reconstructed.

SOLAR ELECTRIC TRAJECTORY/PAYLOAD ANALYSIS

The trajectory data generated in the previous study (Comet Rendezvous Opportunities) was sufficient for purposes of identifying preliminary mission feasibility. However, this data base needed to be expanded for the trade-off analyses required in the present mission study. The main thrust of this new effort is in the area of solar electric propulsion applications. Trade-off parameters of interest include: flight time, arrival date, launch vehicle, powerplant rating and propulsion on-time.

Figure 2-11 illustrates the solar-electric payload capability for missions to Comets Encke and Kopff. Net spacecraft (science and support subsystems) rendezvous weight is plotted as a function of arrival time for the Titan 3C launch vehicle. Points indicated on the curves represent a local optimum flight time and the corresponding optimum power rating for the given arrival time. For example, the 970-day flight to P/Encke arriving 50 days before perihelion delivers 1500 lbs. for a powerplant rating of 16 kw at 1 AU. A general characteristic to be noted is that net spacecraft weight increases as the arrival date approaches perihelion. This increase is more pronounced for the shorter flight times to P/Kopff. Pre-perihelion arrival times (50-100 days) are preferred from a

TABLE 2-14

QUESTIONS TO BE ANSWERED

- WHAT IS THE PHYSICAL STATE AND COMPOSITION OF THE NUCLEUS?
- WHAT IS THE COMPOSITION OF THE MOLECULES AND DUST RELEASED BY THE COMET?
- WHAT IS THE RATE OF RELEASE?
- DOES THE LOSS OF MATERIAL AFFECT THE COMET ORBIT?
- HOW ARE THE OBSERVED COMA MOLECULES AND TAIL IONS FORMED?
- WHAT IS THE CAUSE OF IRREGULARITIES IN THE ACTIVITY OF COMETS?
- HOW DOES A COMET INTERACT WITH THE SOLAR WIND AND SOLAR MAGNETIC FIELD?
- WHAT IS THE SIZE DISTRIBUTION OF DUST LOST FROM THE COMET?

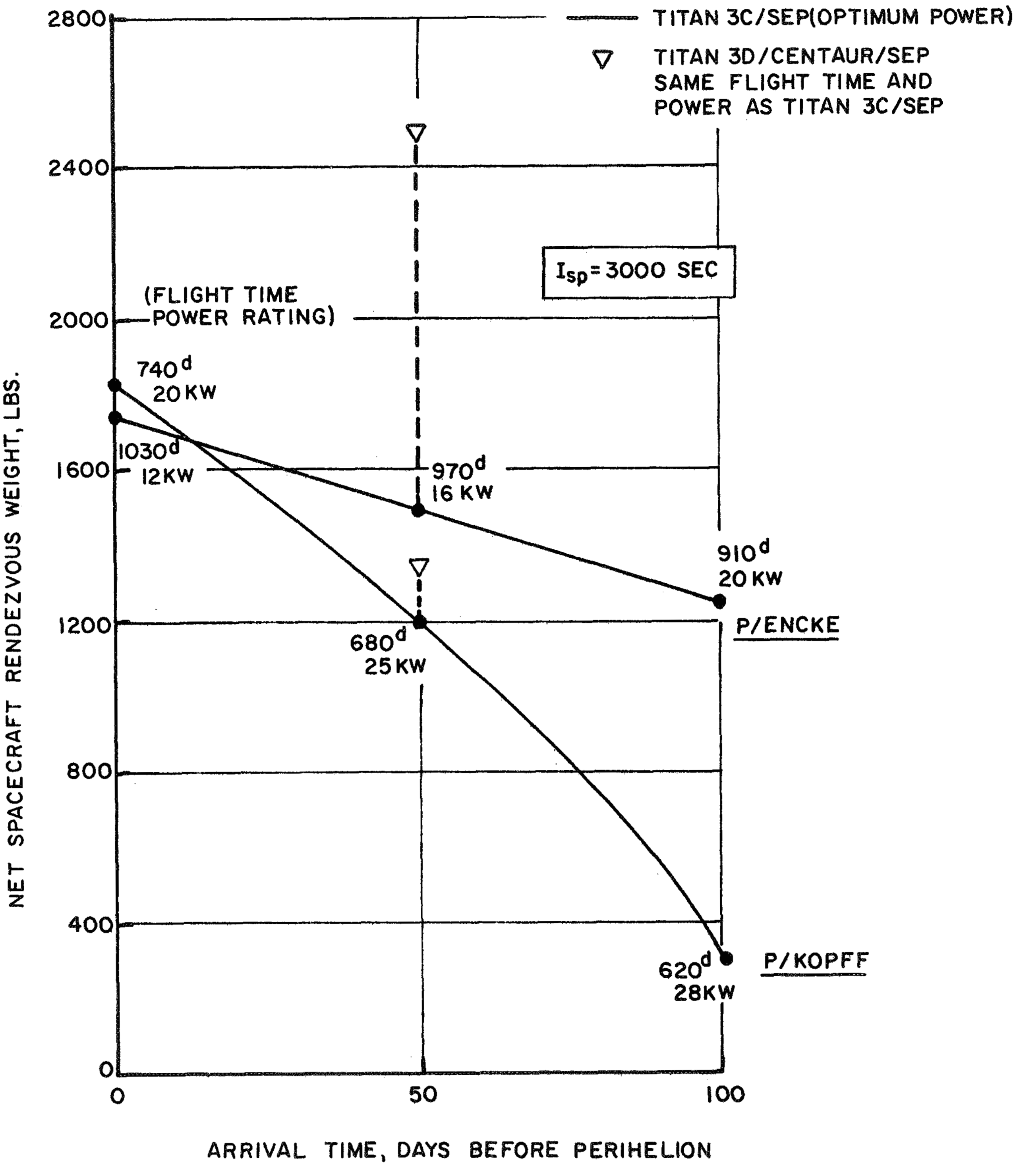


FIGURE 2-11. SOLAR-ELECTRIC PAYLOAD CAPABILITY MISSIONS TO COMETS ENCKE AND KOPFF

science standpoint since the spacecraft would then be in the comet vicinity at the onset of activity. Also, in the case of the 1980 apparition of P/Encke, the comet cannot be observed from earth after 20 days before perihelion. Hence, the trade off between science and trajectory requirements is evident. One notes also that the power rating decreases with later arrivals, and that the power requirements are significantly higher for the shorter flight time trajectories to P/Kopff. In this regard, it should be mentioned that longer flights to P/Kopff (975-1145 days) have been identified; these would alleviate the problem of low payloads and large powerplants. Also shown in the figure is the effect of stepping up to the Titan 3D/Centaur launch vehicle. The increase in payload is quite significant for the Encke mission but much less so for the Kopff mission.

Taking the Encke mission as an example, the excess payload capability of the Titan 3D/Centaur may be utilized effectively to gain engineering advantages in the SEP spacecraft design. Figure 2-12 shows net rendezvous weight as a function of power rating with propulsion on-time as a parameter. The Titan 3D/Centaur capability shown in the previous figure delivered 2500 lbs. at a power rating of 16 kw but required almost full propulsion (~ 970 days). Here we see that both power and propulsion time can be reduced to more desirable levels and still meet the mission payload requirements. For example, assuming a 1200 lb. net spacecraft to be a typical requirement, the propulsion on-time can be decreased to about 600 days for a powerplant rating of 14 kw.

Figure 2-13 illustrates the SEP trajectory profile corresponding to the example design point. The most immediate characteristic of interest is the fact that this trajectory

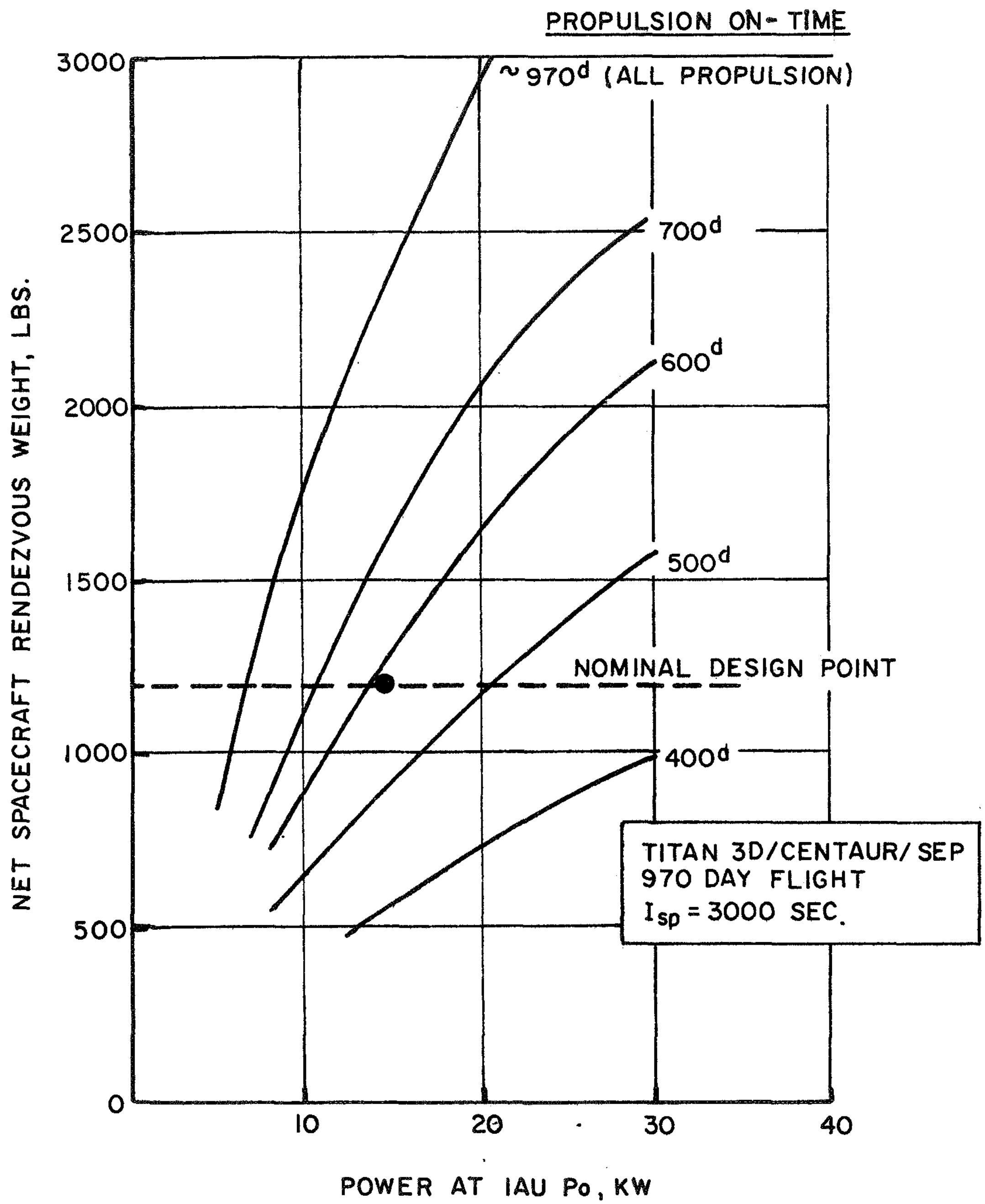


FIGURE 2-12. SOLAR ELECTRIC PAYLOAD CAPABILITY FOR RENDEZVOUS MISSION TO COMET ENCKE (1980) LAUNCH FEB. 15, 1978. ARRIVAL 50 DAYS BEFORE COMET PERIHELION.

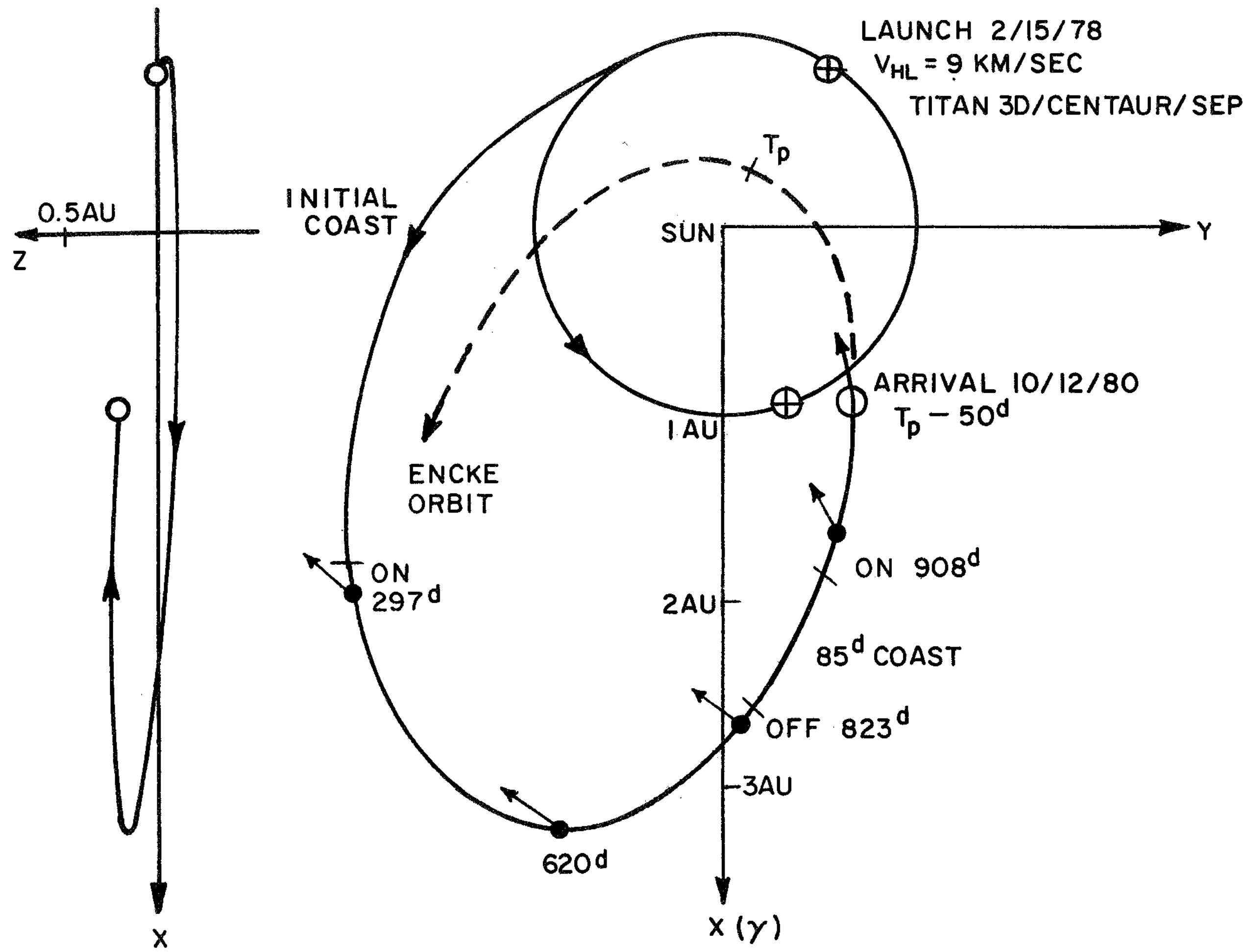


FIGURE 2-13. COMET ENCKE RENDEZVOUS MISSION SOLAR ELECTRIC TRAJECTORY PROFILE (970^d FLIGHT)

calls for an initial coast period of almost 300 days. This comes about because the large hyperbolie launch velocity of 9 km/sec is sufficient to provide the initial trajectory shaping. The low thrust system is most efficiently utilized at a later point (about 3 AU) to turn the trajectory into Encke's orbit and to provide final velocity matching. An 85-day coast period also appears later in the flight profile. The requirement for a long initial coast period is somewhat disconcerting at first glance, but this may not be an operational disadvantage. However, it is clear that Mission Operational Control would want to start-up the thrust subsystem early in the flight, at least for a short time interval, in order to verify or measure system performance. The nominal coast period could be adjusted to account for measured performance deviations.

APPROACH GUIDANCE

Terminal guidance errors to be corrected during the comet approach phase are due to three error sources: (1) spacecraft tracking or orbit determination errors during the interplanetary transit, (2) errors in executing the midcourse velocity impulse (ballistic flight mode) or the thrust program (low thrust flight mode), and (3) comet ephemeris uncertainties. The effect of the first two error sources can be reduced to relatively small values by resorting to intermittant tracking updates and trajectory corrections during the midcourse phase. The limiting accuracy upon comet approach is expected to be due to the third error source. In comparison to the planets, the comet ephemeris (position-in-orbit) errors are quite large. This is due to the relatively short arc near perihelion when comets are observed from earth, and also the non-gravitational accelerations that seem to influence comet motion. Each of the above error sources will be treated in the mission study, but at present the guidance analysis has concentrated on the ephemeris error effect.

Ephemeris errors can be effectively reduced by tracking (earth-based or on-board) the comet during the several months preceeding final rendezvous. Figure 2-14 shows the reduction in miss distance uncertainty for P/Encke assuming an earth-based telescopic observation schedule. The initial uncertainty in this example is 141,000 km and is due mainly to an assumed perihelion time uncertainty of 0.05 days. Recovery is defined as the earliest time when the comet is observable from earth. On the basis of a sighting analysis of the 1980 apparition, recovery is expected to occur about 150 days before perihelion passage. It is seen that the miss uncertainty is reduced to 5500 km after 50 days of tracking and to 600 km after 100 days of tracking (at the nominal arrival date). A late recovery would degrade the orbit determination process during the approach to the comet, although the final accuracy is the same as for the expected recovery. The results shown may be somewhat optimistic in that observations at 2-day intervals are assumed and statistical data processing (least-squares averaging) has been applied assuming a random observation error of 2 arc-sec. If the same observation error consisted mainly of a systematic or bias component, then the miss distance reduction would not be as significant as shown in Figure 2-14.

Figure 2-15 shows the approach guidance ΔV requirements assuming the preceeding earth-based tracking schedule. A two-correction guidance policy is employed. The first correction attempts to null the initial miss of 141,000 km while the second correction attempts to null the residual miss uncertainty existing at the time of the first correction. In the example,

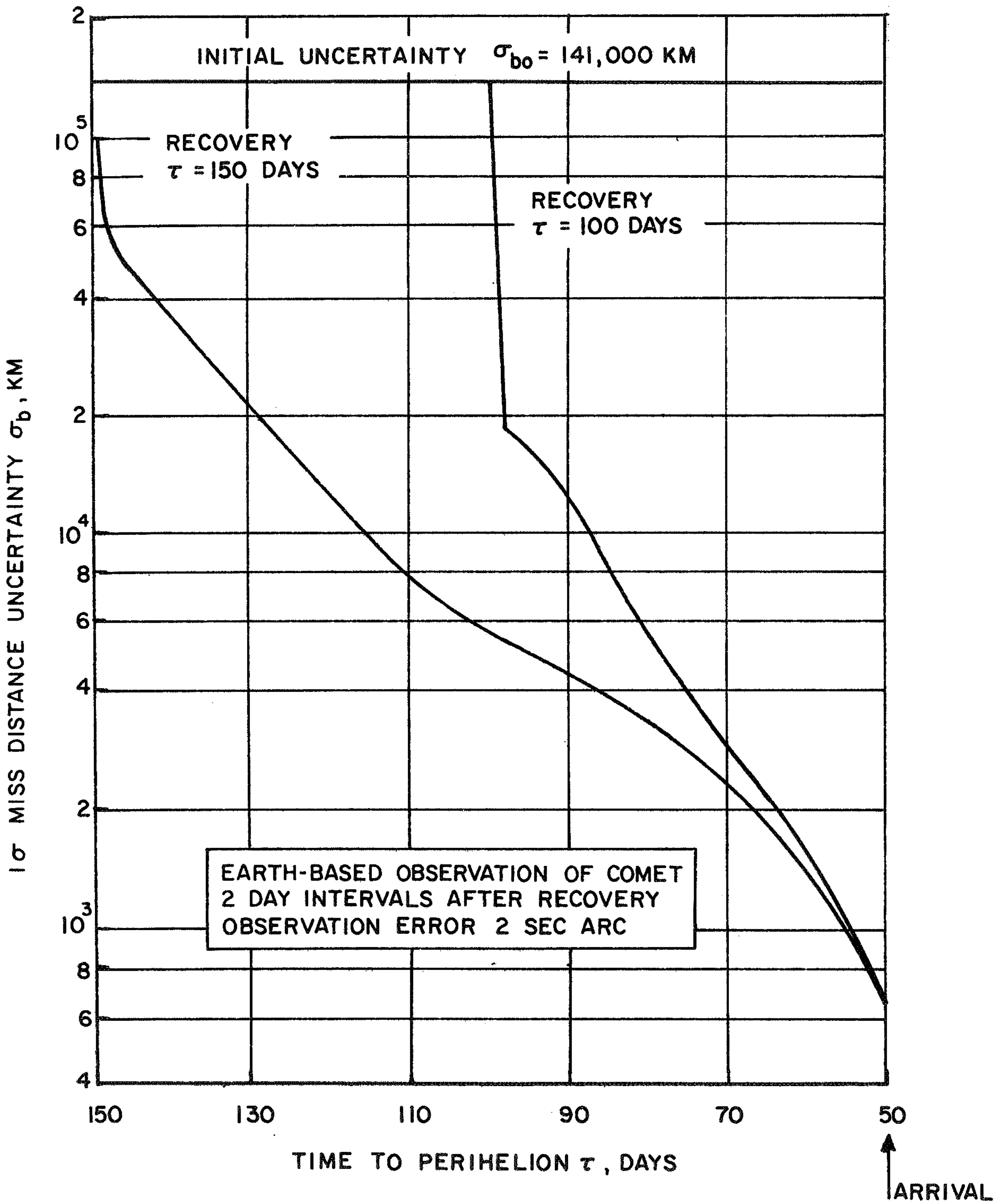


FIGURE 2-14. APPROACH ORBIT DETERMINATION FOR COMET ENCKE/80 RENDEZVOUS MISSION, ARRIVAL 50 DAYS BEFORE PERIHELION.

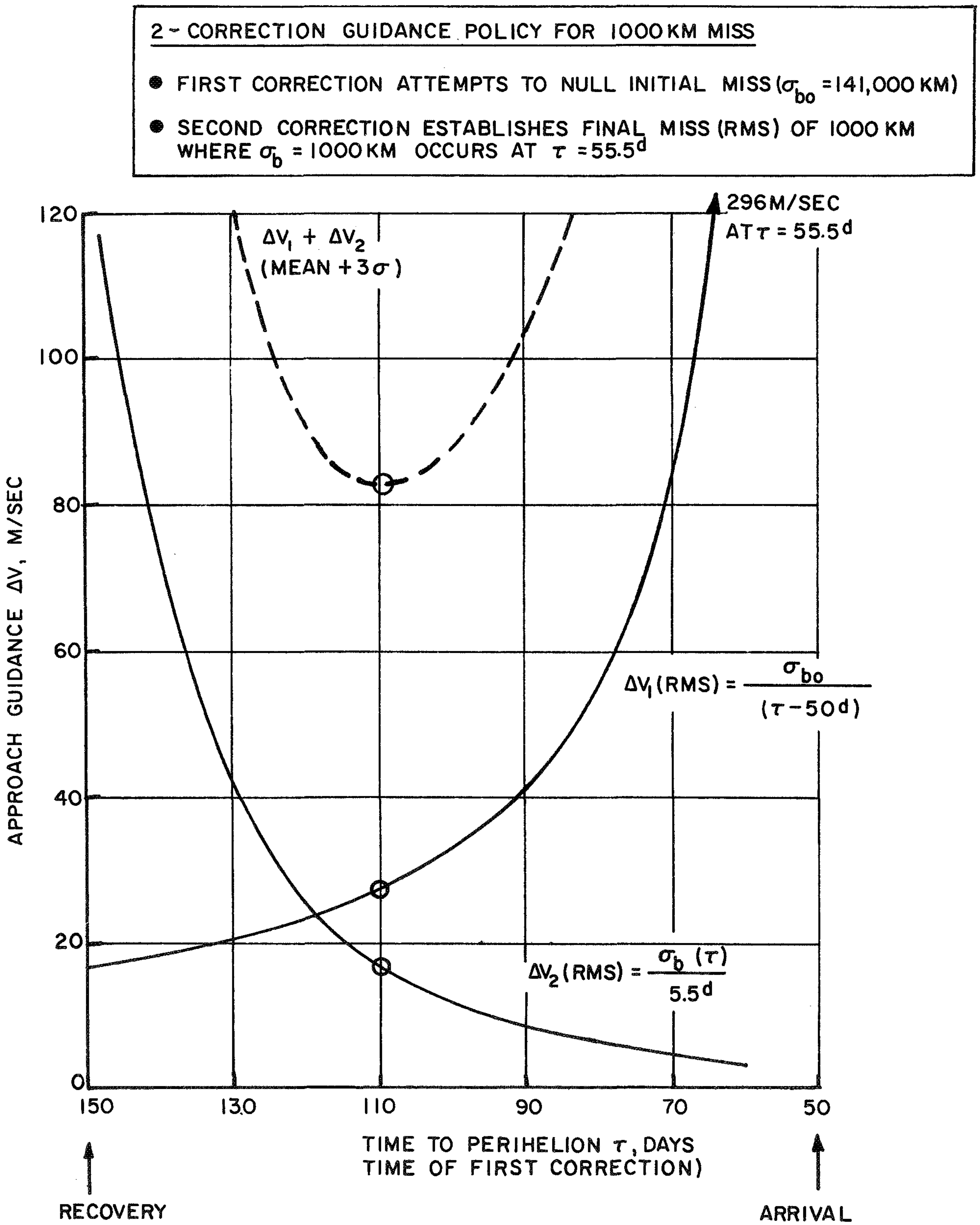


FIGURE 2-15. APPROACH GUIDANCE REQUIREMENTS FOR COMET ENCKE/80 RENDEZVOUS MISSION, ARRIVAL 50 DAYS BEFORE PERIHELION.

the second correction is made 5.5 days before arrival in order to establish a final guidance error of 1000 km. Note that a ΔV of 296 m/sec would be required if only a single approach maneuver were allowed. Optimization of the two-correction policy reduces the RMS ΔV requirement to 27 m/sec (first correction) and 16 m/sec (second correction). A total ΔV budget of 83 m/sec would then yield a 99 percent probability of attaining the final miss of 1000 km.

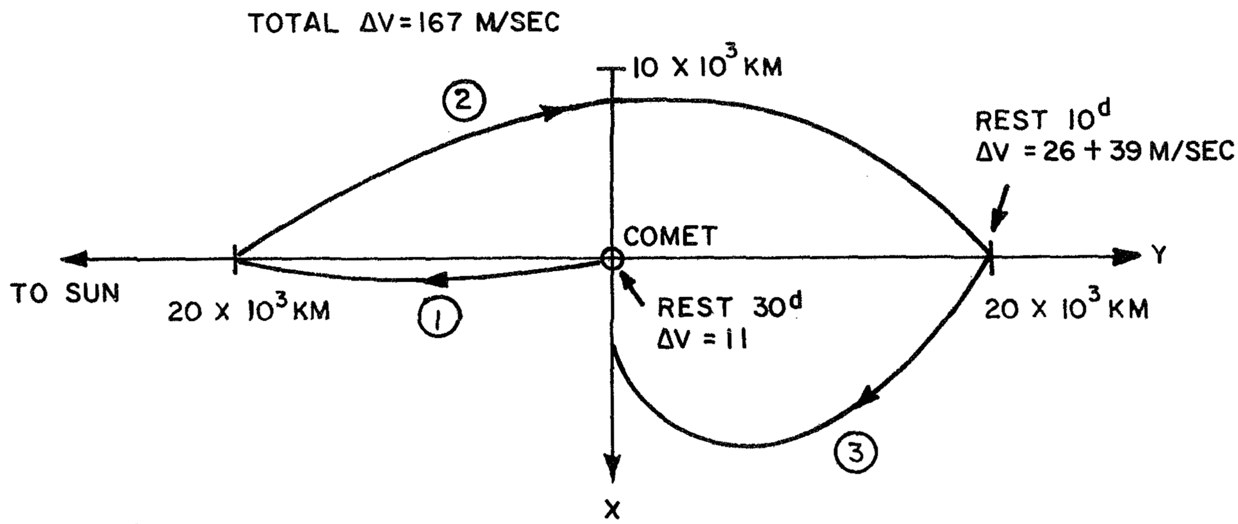
It is desirable that the final error at the rendezvous time be much smaller -- say about 100 km. An on-board comet tracker (e.g., a vidicon system observing the comet against the stellar background) would be required to achieve this level of accuracy. Furthermore, since the on-board tracking uncertainty profile should be significantly better than the earth-based profile over the entire approach phase, the total ΔV requirement can be greatly reduced. Subsequent analysis of the on-board tracking concept will attempt to verify this expected result.

STATIONKEEPING MANEUVERS

The goals of the stationkeeping maneuvers are to allow in-situ measurements of the spatial and temporal development of the comet coma and tail. The remote sensing instruments will provide some information on conditions at other locations. Consider the mission to P/Encke for purposes of an example. Stationkeeping would begin with an investigation of the nucleus from a distance less than 1000 km and over a time interval of about 10 days. Then, at 40 days before perihelion (dbp), the spacecraft should begin a spatial investigation of the coma which will take it 20,000 km away from the nucleus in the sunward direction, arriving 30 dbp. The next station at 10 dbp is on the anti-sunward axis for observations of the tail ions.

While returning to a position near the nucleus (at ~20dbp) the spacecraft will be able to measure dust particles. Because the activity of P/Encke is expected to be greater before perihelion than after, the majority of the translation/stationkeeping maneuvers occurs before perihelion.

Figure 2-16 illustrates the maneuver profile described above and gives a breakdown of the ΔV requirements. These results assume impulsive maneuvers and free-fall translation paths. However, the ΔV requirements would be fairly representative if low thrust maneuvers were employed. It is noted that the total ΔV requirement of 167 m/sec is due largely to the extensive translation maneuvers rather than to stationkeeping at a fixed position. Since the general spatial profile is considered desirable, any necessary reduction in the ΔV budget is perhaps best achieved by reducing the overall dimensions of the translation path. A reduction of the sunline station points to 10,000 km distance would effect a ΔV reduction by approximately one-half, i.e., to 84 m/sec. Finally, Figure 2-17 illustrates an extremely low ΔV (essentially zero) stationkeeping maneuver. This represents a close cirumnavigation of the comet over a 100 day interval centered about perihelion. The dotted line indicates the control-free path due to the differential solar gravitational acceleration acting on the spacecraft by virtue of its displacement from the comet nucleus. The ΔV requirement is essentially proportional to the size of the circumnavigation path. Thus, beginning at a distance of 10,000 km, the ΔV cost would be about 10 m/sec (actually, only 5 m/sec since the final "bring-to-rest" maneuver could be omitted because the mission is completed at this time). While the low cost circumnavigation stationkeeping may be of interest, it is much less desirable from a science standpoint than the profile previously described.



1. $\Delta t = 10^d$, ΔV = 23 M/SEC
2. $\Delta t = 20^d$, ΔV = 45 M/SEC
3. $\Delta t = 20^d$, ΔV = 23 M/SEC

FIGURE 2-16. TRANSLATION/STATION KEEPING MANEUVERS NEAR COMET ENCKE

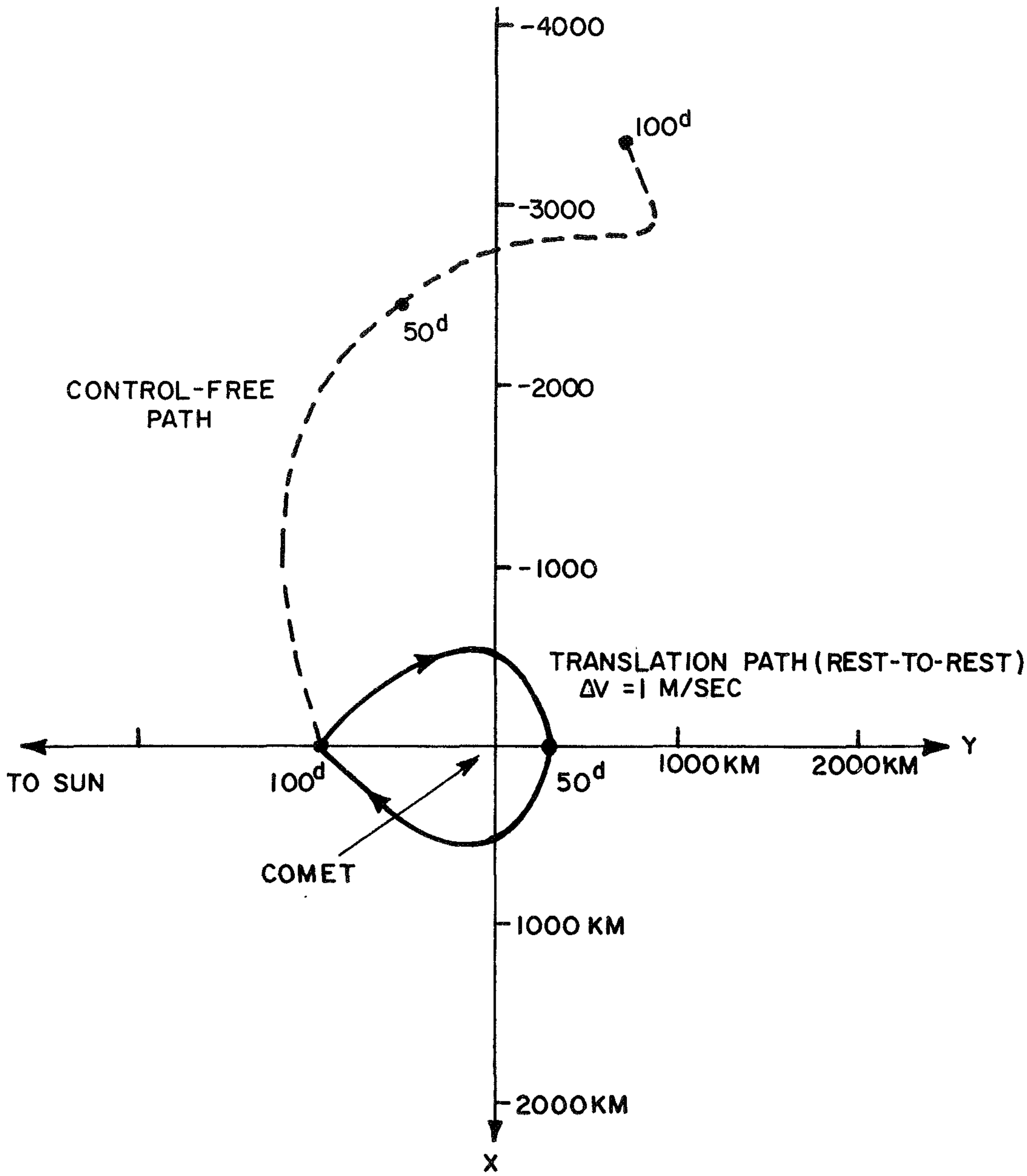


FIGURE 2-17. CLOSE CIRCUMNAVIGATION ABOUT COMET ENCKE/80
100 DAY INTERVAL FROM $T_p - 50^d$ TO $T_p + 50^d$

2.3 TRAJECTORY STUDIES

Report No. T-23
"MODIFICATION OF AN OPTIMUM MULTIPLE IMPULSE COMPUTER PROGRAM FOR HYBRID TRAJECTORY OPTIMIZATION"
J. I. Waters
May 1970

Hybrid low-thrust/gravity-assist trajectories have been proposed as a means of further reducing flight times and increasing payloads over those obtained when either propulsion mode is used separately. The objective of this study was to find an expedient way to develop a hybrid trajectory optimization capability.

The chosen trajectory optimization method finds optimum impulsive trajectories consisting of an arbitrary number of velocity impulse points connected by conic free fall arcs. Optimization is accomplished by a standard conjugate gradient search routine which operates on the impulse times and position components in order to minimize the following performing index:

$$J = (1-\xi)\ (t_K - t_1) - \xi \sum_{k=1}^{K} \Delta m_k$$

$$+ \sum_{k=1}^{K-1} \int_{t_k}^{t_{k+1}} \left[\underline{\lambda}^T (\underline{\dot{v}} - \underline{f}(x)) + \underline{\mu}^T (\underline{\dot{x}} - \underline{v}) \right] dt,$$

where K is the number of impulse points. ξ can be set at different values ($0 < \xi \leq 1$) to provide tradeoffs between minimum time of flight and minimum mass expenditure.

Constraint options can be applied at any impulse point. These include departure from and capture into planetary orbit and rendezvous or flyby at a planet or arbitrary body. Gravity-assisted turns are approximated by forcing a flyby constraint at the planet and using an extremely inefficient propulsion system to provide that part of the flyby impulse which is not available from the swingby. The optimization process will then move the planetary encounter time until only a negligible impulse is required from the hypothetical propulsion system.

Low-thrust segments of the hybrid trajectories are approximated by closely and evenly spaced impulse points. Constant-thrust and solar-powered cases are approximated by computing an available impulse area for each point as a function of propulsion parameters, current mass, solar radius and time between impulses. This impulse area is used as a limit to constrain the corresponding impulse value in a fashion similar to that which was outlined above for the gravity swingby constraint.

It is expected that a hybrid trajectory optimization capability can be developed in a relatively short time. An existing Astro Sciences impulsive trajectory optimization program can be modified in accordance with the following schedule:

STEP	COMMENTS
1. Add N-impulse capability.	Requires redimensioning of storage arrays.
2. Convert to minimum mass loss formulation.	Present version is minimum total $\lvert\Delta V\rvert$. Includes mass computation.
3. Expand constraint options to any point.	Constraints allowed only at end points in present version.

STEP	COMMENTS
4. Add gravity-turn limiting constraint.	To be available at any point and apply at any planet.
5. Add uniform time increment capability.	Arbitrary groups of successive impulse times would be effected.
6. Add constant thrust and solar powered limiting constraints on impulses.	

Based upon our past experience, the resulting program can be expected to exhibit faster and much more reliable convergence characteristics than programs using numerical integration and a Newton-Raphson search on the initial multiplier values.

Report No. T-25
"TRAJECTORY AND PROPULSION CHARACTERISTICS OF COMET RENDEZVOUS OPPORTUNITIES"
A. L. Friedlander, J. C. Niehoff, and J. I. Waters
August 1970

This report presents a new look at spaceflight mission opportunities to the comets in the time period 1975-2000. Previous studies of comet missions have been restricted mainly to the flyby trajectory mode. Although offering short flight times and low launch velocity requirements, comet flybys suffer from the standpoint of scientific information return because of the very high flyby velocities. Current interest is now centered on the rendezvous (orbit matching) mission which allows the spacecraft many months to monitor the variations in physical activity as the comet approaches and passes through perihelion.

This report expands upon earlier work in this area (Friedlander, Niehoff and Waters, 1969) in terms of the scope of mission opportunities available and the comparison of candidate flight modes for performing these missions. Specifically, the objective is to identify promising rendezvous mission opportunities and flight modes in the time period 1975-2000 from the standpoint of trajectory requirements and launch vehicle/payload capabilities. Two ballistic and two low-thrust flight modes are considered. Ballistic flight modes include: (1) direct transfer utilizing three or more velocity impulses, and (2) gravity-assist transfers via the planet Jupiter thus eliminating the midcourse propulsive impulse. The low-thrust propulsion modes include application of: (1) nuclear-electric powerplants, and (2) solar-electric powerplants. All trajectories are optimized to effectively minimize payload (net spacecraft mass delivered) for flight time and launch vehicle selections. Emphasis is placed on programmed Titan-class launch vehicle and flight times consistent with delivering a payload of about 1000 pounds. An additional constraint generally applied is that the rendezvous point occur in the region 0-200 days before comet perihelion.

Comet mission opportunities have been selected initially on the basis of special scientific interest and Earth-based sighting criteria. The sighting criteria refer to the recovery of the comet by telescopic observation from Earth prior to the time of rendezvous and sufficient brightness afterwards for obtaining spectroscopic measurements. An early recovery provides an accurate update of the comet's position in orbit, thereby easing the spacecraft guidance problem. Spectroscopic measurements made from Earth are considered for purposes of correlating spacecraft measurements. Although they are thought to be important, the sighting criteria are not necessarily hard constraints dictating mission success value. Fortunately many mission opportunities do satisfy the sighting criteria. Table 2-15 lists those comet apparitions which satisfy the sighting criteria. Also, those flight modes for which successful missions were found are noted in the last four columns. Note that the nuclear propulsion portion of the study has been devoted almost completely to a study of the Halley/86 mission.

An attractive early mission opportunity to Comet Encke, a well known short period comet, in 1980, has been found. Figure 2-18 shows a performance comparison of the 3-impulse ballistic and solar-electric flight modes and the nuclear-electric flight mode (1990 apparition). It is noted that the 1980 and 1990 apparitions have comparable trajectory characteristics for either flight mode because the orbital period of

TABLE 2-15

SUMMARY OF COMET RENDEZVOUS STUDY

		BALLISTIC MODES *		LOW THRUST MODES *	
YEAR OF APPARITION	COMETS INVESTIGATED	MULTIPLE IMPULSE	JUPITER GRAVITY ASSIST	NUCLEAR ELECTRIC	SOLAR ELECTRIC
80	ENCKE	0		X (90)**	X
82	d'ARREST	X	X		X
82	GRIGG-SKJELLERUP	0			
83	KOPFF	X			X
84	ENCKE	0			
85	GIACOBINI-ZINNER	0	0		
86	HALLEY		0	X	0
87	BORRELLY	0	0		
88	TEMPLE-2	X			
91	FAYE	0			
93	FORBES	X			
93	SCHAUMASSE	0			
94	TUTTLE	0			
95	PERRINE-MRKOS	0			
96	KOPFF	X			
98	GIACOBINI-ZINNER	0			

* X indicates promising mission found

0 indicates no promising mission found

A blank indicates flight mode was not considered in this study

** 1990 opportunity (similar in geometry to 1980 opportunity)

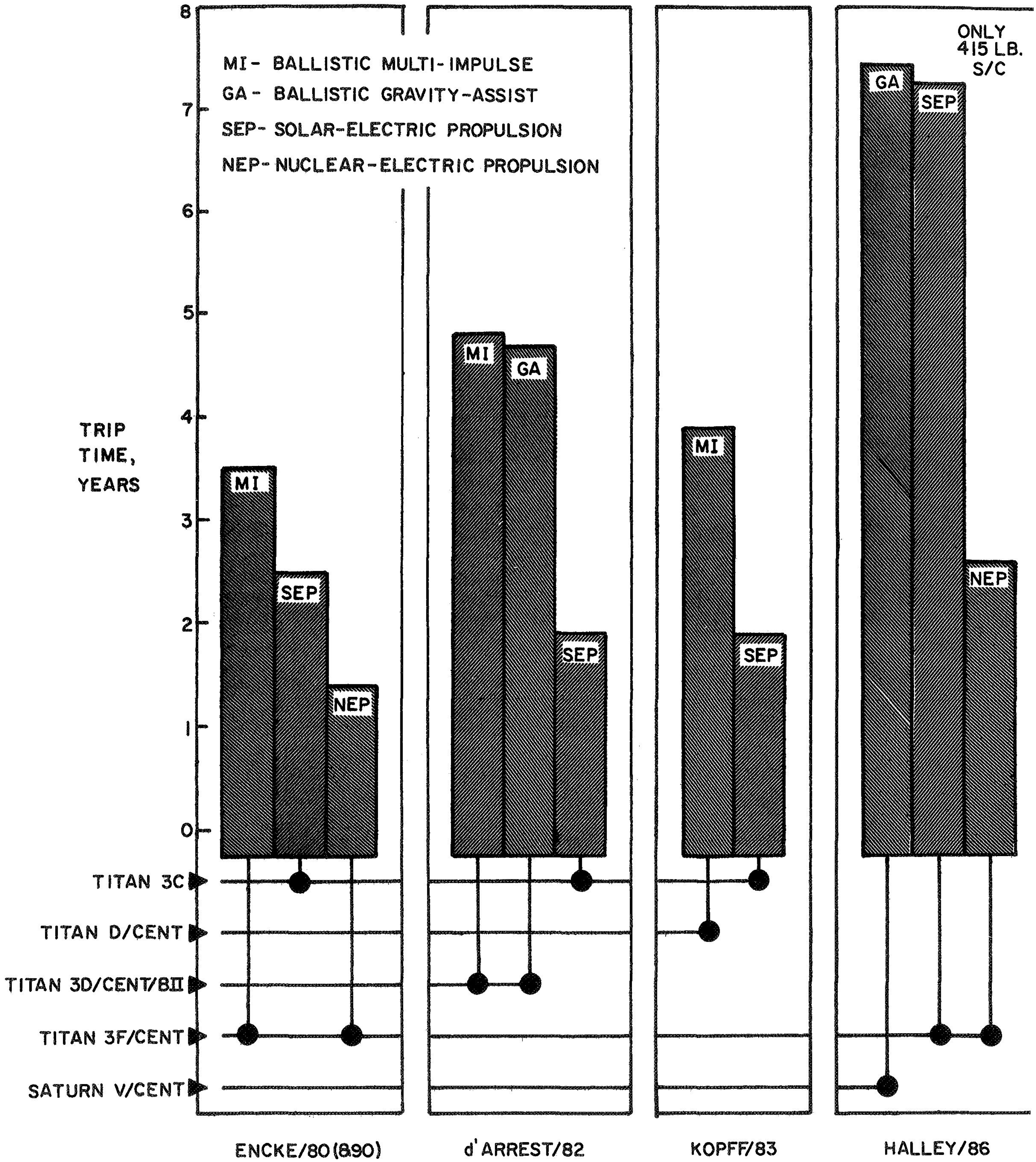


FIGURE 2-18. COMET FLIGHT MODE COMPARISONS FOR ∿ 1000 LB. RENDEZVOUS SPACECRAFT

Comet Encke is about 3.3 years. Ballistic payload capability is 820-1000 pounds for the Titan 3D/Centaur or Titan 3F/Centaur launch vehicles. A nuclear-electric spacecraft launched by the same Titan class vehicle offers the advantages of a two year reduction in flight time and a larger payload. A solar-electric spacecraft is also attractive, requiring a Titan 3C launch vehicle and 2.5 years of flight time to deliver 1000 pounds. The Encke/80 mission would provide an early opportunity to develop comet rendezvous technology prior to the arrival of Halley's Comet in 1985.

The d'Arrest/82 mission is characterized by the relatively long flight times required by the ballistic flight modes. The solar-electric mode is much more effective, requiring only a 1.9 year flight time. Also included in Figure 2-18 are performance comparisons of ballistic and solar-electric missions to Kopff/83, the details of which are quite similar to those of the d'Arrest/82 missions. The desirability of missions to these two comets would seem to depend upon the interest of the scientific community in utilizing these opportunities.

The most outstanding comet mission from the standpoint of scientific and public interest is that to Halley's Comet which is due to return in 1985-86. A rendezvous with Halley is especially difficult because of the unique retrograde feature of its orbital motion. Figure 2-18 compares the performance characteristics of the ballistic and low-thrust flight modes in achieving a Halley rendezvous. The ballistic mode uses gravity-assist via a Jupiter swingby. In order to deliver a payload of about 1000 pounds ballistically, a Saturn V/Centaur launch vehicle is required and the flight time is almost 8 years. The nuclear-electric spacecraft launched by the smaller Titan 3F/Centaur can deliver a payload in excess of 1000 pounds, and requires a flight time of only 2.6 years. An alternative might be the solar-electric Halley mission, which does not require a nuclear-electric system or a Saturn V launch vehicle but can deliver only 415 pounds with a 7.5 year flight time.

Effective accomplishment of the mission to Halley's Comet would seem to depend upon the development and availability of nuclear-electric propulsion by 1983. At this writing, there is still the possibility that nuclear-electric propulsion would be available in time for the Halley mission or that a commitment to a large ballistic launch vehicle could be made if given an early priority. Two other alternatives for Halley rendezvous yet to be studied are: (1) a solar-electric powered spacecraft employing a large solar collector in order to maintain high power levels throughout the flight, and (2) a combined Jupiter-assisted solar-electric mission. If it should turn out that Halley rendezvous is completely impractical, an alternative mission mode might be multiple intercept probes arriving at different points of the perihelion passage.

Certain tentative generalizations concerning comet rendezvous missions are demonstrated in Figure 2-18 and were found to apply over the entire group of missions considered. Remembering that Comet Halley is a unique case, we may conclude that:

(1) Ballistic comet rendezvous missions will typically require upwards of four years of flight time and advanced Titan/Centaur launch vehicles,

(2) There are no attractive missions for which the Jupiter gravity-assist technique can significantly improve upon the impulsive ballistic flight mode. Gravity-assisted rendezvous trajectories do, however, provide better payload

performance at much earlier arrival dates (300-500 days before perihelion) which may prove to be a mission advantage after further study of arrival date effects on mission success.

(3) Solar-electric propulsion can reduce flight times from the four years typically required by ballistic flights to about two years.

(4) While marginally effective missions to Halley/86 are possible with solar-electric propulsion or Jupiter gravity-assist, the availablility of nuclear-electric propulsion would result in a significant improvement in trip time over these flight modes.

Comet rendezvous missions in the time period 1975-2000 are both attractive and feasible from a trajectory/payload standpoint. Several mission profiles utilizing near state-of-the art ballistic flight systems have been identified. The superior performance potential of future nuclear-electric spacecraft has been demonstrated for the Halley mission opportunity. Significant performance improvement can be obtained by using solar-electric propulsion for comet rendezvous missions with the possible exception of Halley/86. An extension of the present study is necessary to complete the picture of comet rendezvous as a class of missions. Subject areas of particular importance to complete mission definition include science objectives, experiment design, transfer and approach guidance, and stationkeeping maneuvers.

2.4 SPACECRAFT TECHNOLOGY

Technical Memorandum No. S-7
"RADAR EXPLORATION OF VENUS"
By D. L. Roberts and H. J. Goldman
June 1970.

This memorandum is the result of a short study of spacecraft radar systems suitable for the Planetary Explorer class of Venus Mapping Orbiters. It attempts to place in perspective the different modes of radar operation, to compare their respective contributions and identify their major system requirements. The details of the radar system finally recommended were included in the submissions of the Goddard Space Flight Center to the Space Science Board of The National Academy of Sciences in June 1970.

Earth-based radar measurements of Venus provide an indication of the conditions a spacecraft radar system must be designed to meet. Besides determining the rotation rate and diameter of Venus earth-based radar has provided some information on the surface. The surface is fairly smooth with a few pronounced surface features or rough patches. The average slope is about 7° or 8° compared with 10° for the moon. There are clear differences in the reflectivity of Venus as a function of wavelength, going from 20% at 6m down to about 1% at 3.6 cm. At long wavelengths the high radar cross section and the dielectric constant (ϵ=3.5-5) implies that the surface is covered with solid rocks, possibly silicates, rather than a regolith or large expanses of liquids.

The major problems with earth-based radar measurements is the limited spatial resolution obtainable. Current systems provide about a 100 km resolution limited to regions within ± 30° of the Venusian equator on the earth-facing hemisphere. Projected improvements will allow resolutions between 1 and 5 km within ± 10° of the sub-earth point at conjunction. Only orbiting spacecraft radar systems have the potential of providing this resolution over the entire planet.

It is not known at present whether the reflection properties of Venus' surface will be predominantly specular or diffuse at the wavelengths considered for spacecraft radar (13 cm). Specular reflection can be likened to the reflection from a dirty mirror whereas diffuse reflection is like that from a sheet of white paper. It is necessary to consider the consequences of each of these cases for spacecraft radar. The principle effects are summarized below:

BASIC PROPERTIES OF RADAR REFLECTION

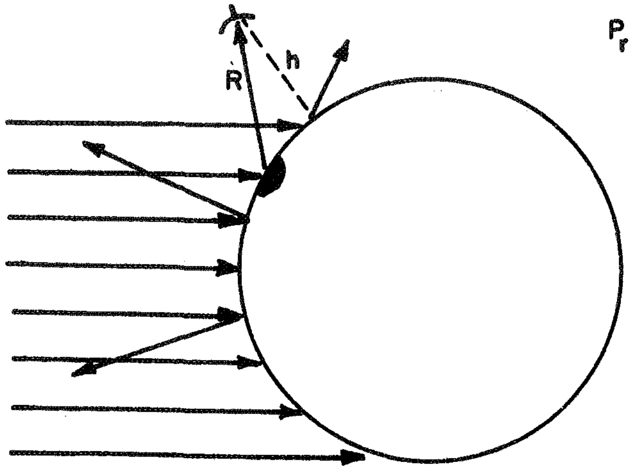
SPECULAR	DIFFUSE
ONLY AT SPECIFIC ANGLES OF INCIDENCE AND REFLECTION.	ANY LOOK ANGLE INCLUDING SIDE LOOKING.
SIGNAL STRENGTH INDEPENDENT OF FIELD OF VIEW (ANTENNA BEAM WIDTH).	SIGNAL STRENGTH DEPENDS ON AREA VIEWED.
RESOLUTION DEFINED BY REFLECTING SURFACE.	RESOLUTION DEFINED BY SIGNAL PROCESSING (DOPPLER OR TIME).
CHARACTERISTIC POLARIZATION	RANDOM POLARIZATION.

The most important differences between specular and diffuse reflection show up in the basic radar equation. These are summarized in Figures 2-19 and 2-20. (Note that side looking radar can be used only with diffuse reflection).

Bistatic radar operates best when the signal is transmitted from Earth and received on the spacecraft. If the planet is diffusely reflecting at 13 cm the bistatic system will give no information on a scale of 200 km. If the planet is specularly reflecting, however, a bistatic system employing a 50 lb. receiving system on board a spacecraft may obtain a 200 km resolution, although coverage is limited to about 70% of the planet. The complex geometry of the specular point with respect to the spacecraft makes the data interpretation extremely difficult.

Monostatic radar on the spacecraft offers the most in terms of resolution, coverage and range of useable altitudes. Table 2-16 gives some detailed characteristics of the vertical incidence radar system selected for inclusion as part of the Planetary Explorer Mapping Orbiter payload. It is designed to provide useful data regardless of the reflecting properties of Venus' surface. If the reflection is predominately specular the system can be used throughout the entire 400 x 50,000 km orbit. If diffuse, measurements can be made over 90 degrees of the orbit with restricted coverage.

BISTATIC



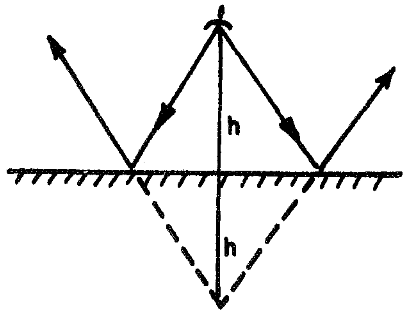
$$P_r = \frac{P_t\, G_t}{4\pi R^2_{E-V}} \cdot \frac{\sigma_B |\rho|^2}{4\pi (R_v + h)^2} \cdot \frac{\lambda^2}{4\pi}$$

ASSUMPTIONS:

APPARENT SOURCE IS WHOLE AREA OF PLANET
$\sigma_B = \pi R_v^2$ ACTING AT CENTER OF PLANET
RESOLUTION $\sim \alpha R$
REFLECTION COEFF $\sim$ ·14
OMNI ANTENNA, G_r = 0 db
DOPPLER FILTERS TO LOCATE SPECULAR POINT

MONOSTATIC
($h \ll R_v$)

$$P_r = \frac{P_t\, G_t}{4\pi (2h)^2} \cdot |\rho|^2 \cdot \frac{G_r \lambda^2}{4\pi}$$

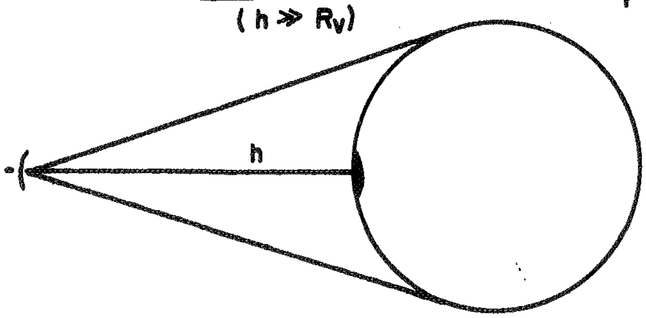


ASSUMPTIONS:

FLAT MIRROR REFLECTING SURFACE,
APPARENT SOURCE 2h FROM RECEIVER.
RESOLUTION $\sim \alpha h$
$G_r = G_t$

MONOSTATIC
($h \gg R_v$)

$$P_r = \frac{P_t\, G_t}{4\pi (R_v/2 + h)^2} \cdot \frac{\pi R_v^2 |\rho|^2}{4\pi h^2} \cdot \frac{G_r \lambda^2}{4\pi}$$



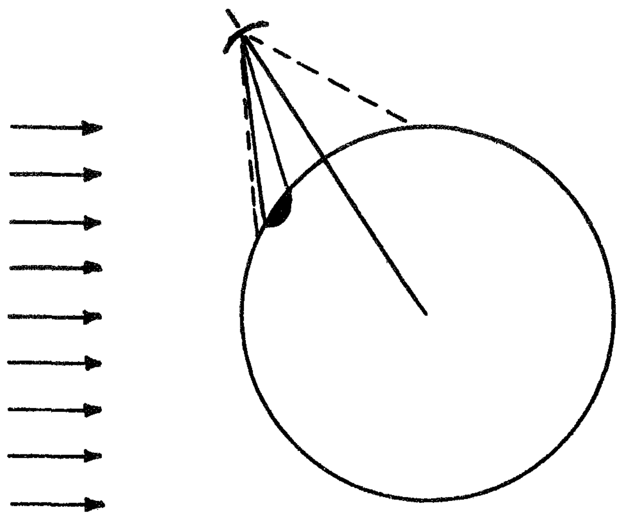
ASSUMPTIONS:

APPARENT SOURCE IS WHOLE AREA OF
PLANET πR_v^2 AT $R_v/2$ BENEATH SURFACE
RESOLUTION $\sim \alpha h$
$G_r = G_t$

FIGURE 2-19. BASIC RADAR EQUATIONS; SPECULAR REFLECTION

BISTATIC

$$P_r = \frac{P_t G_t}{4\pi R^2_{E-V}} \cdot \frac{\sigma |\rho|^2}{4\pi h^2} \cdot \frac{\lambda^2}{4\pi}$$

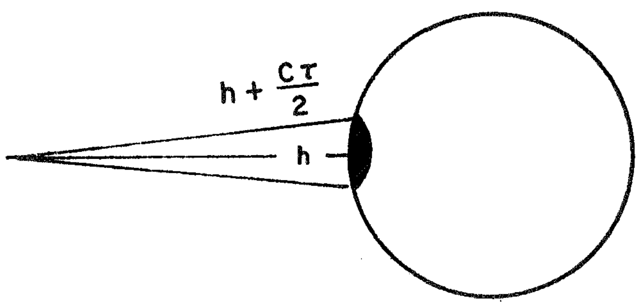


ASSUMPTIONS:

DOPPLER FILTERS TO LOCATE AREAS $\sim 200\ KM^2$
CROSS SECT. $\sigma = \pi/4\ (200)^2\ KM^2$
REFLECTION COEF $|\rho|^2 \sim .10$

MONOSTATIC

$$P_r = \frac{P_t G_t}{4\pi h^2} \cdot \frac{|\rho|^2 (\pi c \tau h)}{4\pi h^2} \cdot \frac{G_r \lambda^2}{4\pi}$$



ASSUMPTIONS:

PULSE WIDTH LIMITED $RES^N = 2\sqrt{c\tau h}$
NO DOPPLER FILTERS
TIME GATES FOR ANALYSIS

FIGURE 2-20. BASIC RADAR EQUATIONS; DIFFUSE REFLECTION

TABLE 2-16

RADAR FOR REGIONAL MEASUREMENTS (1-20 KM)

(RECOMMENDED FOR PLANETARY EXPLORER)

CHARACTERISTIC	VERTICAL INCIDENCE	
	SPECULAR	DIFFUSE
Basic System	Pulse τ = 300 ns	Pulse τ = 300 ns
Antenna Gain, db	28.5 (5' dia steerable)	28.5 (5' dia steerable)
Power, Watts	10 Kw Peak	10 Kw Peak
Signal, dbw	-112 @ 1500 Km	- 141 @ 1500 Km
	-130 @ 50,000 Km	-168 @ 50,000 Km
Noise, dbw	-133	-133
S/N	+21 @ 1500 Km	-8 @ 1500 Km
	+3 @ 50,000 Km	-35 @ 50,000 Km
Maximum Altitude, km	50,000	1,500
Resolution	5 Km	20 Km
Coverage	100% in 120 days	40% in 240 days
Approx. Weight, lbs.	92	92
Approx. Power, Watts	60	60

Report No. S-8

"THE PLANETARY EXPLORATION POTENTIALS OF SPACECRAFT RADAR"

by H. Goldman and R. Brandenburg

October 1970.

This report discusses the characteristics, current status, and potentials of earth-based and spacecraft radar for planetary exploration. Radar sensors in general have several distinct advantages over most other instruments designed for the study of planetary surfaces, particularly visual imagers. Radar functions independently of naturally occurring electromagnetic flux at wavelengths only slightly affected by dense atmospheres or cloud cover. This allows the radar sensor to topographically map the cloud covered or night side surface of a planet, while a visual imager is restricted to the clear, day side.

Earth-based radar systems are able to provide large-scale information on the characteristics of the nearby terrestrial planets; Mercury, Venus and Mars. Future improvements on these systems will allow the resolution of surface features down to several tens of kilometers in size. However, because earth-based systems are approaching practical limits in size, power, and noise reduction, capabilities providing resolutions as fine as hundreds of meters are not foreseeable.

Jupiter and the outer planets have not yet been reliably detected, but preliminary observations indicate that because of extremely low reflectivities earth-based radar systems may not be able to provide much information on these planets. With the projected improvements of earth-based radar sites these planets will only be observable at opposition, and even then will not be able to be studied very thoroughly. The poor resolution of the terrestrial planets and the possible non-detectability of the outer planets by earth-based systems points to the need for spacecraft radar systems.

Spacecraft radar systems may either operate alone or in conjunction with an earth-based radar site. If an earth-based terminal and a spacecraft terminal are to be used (possibly to minimize the spacecraft radar system's weight and power requirements) the optimum operation is to use the earth-based site as a transmitter and the spacecraft as a receiver. This mode has the additional advantage of allowing the earth-based receiver to simultaneously study the planet's gross characteristics in the same manner as current earth-based systems. If there is a choice between a bistatic and a monostatic operation, using one spacecraft system as both transmitter and receiver, calculations show the monostatic system to be the more favorable at the terrestrial planets. For Jupiter radar studies there is a trade-off between two factors; antenna size and spacecraft altitude. For a fixed spacecraft altitude there exists a critical antenna size. If the spacecraft radar antenna is smaller than this critical size the bistatic mode, using an earth-based transmitter, gives a higher final signal to noise ratio than the monostatic mode. If the spacecraft's antenna is larger than this critical size the monostatic system is superior. For a spacecraft altitude of $\sim$ 2.8 Jupiter radii the critical antenna size is $\sim$ 10 feet. As the altitude increases the critical size increases approaching 100 feet in diameter at altitudes near thirty Jupiter radii.

Of the several types of monostatic radar available, the non-coherent sidelooking type is probably the most advanced system compatible with current spacecraft design. Power weight and data requirements do not pose any significant problems in the use of this type of radar for mapping the terrestrial planets. Radar systems weighing about 50 kg (including data recorder), with antennas approximately 1 x 15 meters in size, and requiring about 25 watts of input power are able to provide surface resolutions of about 1 x 20 meters. Processed data rates for these for these systems run at several thousand bits per second, while unprocessed rates are about 10^5 bps.

For Jupiter, however, spacecraft-borne radar mapping systems do not appear to be feasible unless significant advancements are made over the current state-of-art. Even for resolutions as poor as 100 km, the radar antenna for a non-coherent mapping system must be about two thousand wavelengths long (based on altitudes consistent with current studies of Jupiter flyby and orbiter missions). Such an antenna exceeds the foreseeable state-of-art by at least a factor of two, in terms of dimensional tolerances. Furthermore, at an operating wavelength of ten cm (the minimum wavelength providing penetration of the "clear" upper atmosphere) the antenna would be 200 meters in length. Estimates of weight and power consumption indicate that the requirements for a Jupiter radar mapping system are likely to be an order of magnitude more severe than for the terrestrial planets. Use of synthetic aperture systems would minimize the antenna difficulties, but the number of pulses per resolution element which must be processed is likely to create serious, if not insurmountable, data storage and transmission problems.

3. SPECIAL STUDIES AND TECHNICAL NOTES

3. SPECIAL STUDIES AND TECHNICAL NOTES

3.1 Special Studies

Launch Vehicle Selection for Outer Planets Mission

J. C. Niehoff

This short-study task was completed during the first quarter of the past contract year. Initially its purpose was to provide quantitative justification for the seven-segment version of the Titan III/Centaur launch vehicle. However, a fair analysis requires comparing the capabilities of a number of launch vehicle-upper stage combinations. The final list of launch vehicle configurations selected for consideration included:

1) Titan IIID/Centaur/Burner II
2) Titan IIIF/Centaur
3) Titan IIIF/Centaur/Burner II
4) Titan IIID/Centaur/Solar-Electric
5) Titan IIIF/Centaur/Solar-Electric
6) Titan IIIF/Centaur I/Kick
7) SIC/SIVB/Centaur
8) Titan IIIF/Centaur/Nuclear-Electric
9) SIC/SIVB/Centaur/Solar-Electric
10) SIC/SIVB/Nuclear-Electric.

A comprehensive analysis of launch vehicle requirements also depends on a list of mission representative of an outer planets exploration program. For this, the 1969 PEPP plan for outer planet missions was extended to 1990 on the basis of one mission per year. The resulting mission plan is presented in Table 3-1. The 12 outer planet missions shown are scheduled so

TABLE 3-1

EXTENDED OUTER PLANETS MISSION PLAN

	No.	77	78	79	80	81	82	83	84	85	86	87	88	89
Swingby Missions														
Jupiter-Saturn-Pluto	1	X[1]	O(J)	O(S)					O(P)					
Jupiter (Pb)[2]-Saturn	2		X		O(J)		O(S)							
Jupiter-Uranus-Neptune	3			X		O(J)		O(U)			O(N)			
Saturn (Pb)-Uranus (Pb)-Neptune	5					X			O(S)			O(U)		
Orbiter Missions														
Jupiter	4				X			O				O		
Saturn	6						X							
Jupiter (Pb)	7								X		O			
Uranus	8									X				
Neptune	10											X		
Saturn (Pb)	11												X	
Jupiter (Pb)	12													X
Lander Missions														
Jupiter Satellite	9										X			O
Rendezvous Missions														
Comet Halley[3]								X		O				

1. Symbol key: X - Launch, O - Planet Arrival
2. (Pb) indicates atmospheric probes are included
3. Comet Halley rendezvous mission included as a time-line for introduction of nuclear-electric propulsion.

that collected data is in hand at least two years prior to the launch of a succeeding mission to the same planet, i.e., this is a sequential mission-type plan.

A number of constraints and assumptions were made to generate a consistant set of propulsion requirements for the 12 outer planet missions. The base spacecraft weight was set at 1500 lbs. This excludes retro propulsion for planetary orbiters. When atmosphere probes were used they were assumed to have a total weight of 500 lbs. and were added in pairs, i.e., 1000 lbs. Conditions over a 10-day window were considered in determining launch energy requirements. For planetary capture a space-storable propulsion system with an I_{sp} of 385 sec was assumed. The orbit period was constrained to 15 days at all planets, while the periapse radii were selected as follows:

Jupiter - 3 planet radii
Saturn - 3 planet radii
Uranus - 2 planet radii
Neptune - 2 planet radii.

Mission accomplishments, in terms of flight time, with each launch vehicle combination listed above were compared for each of the twelve missions. The comparisons can be summarized by answering the following two questions:

a) How effective would the seven-segment Titan be in reducing flight time and adding new missions to an outer planets program?

b) What additional increase in launch vehicle capability is required to complete all missions given in the extended plan in Table 3-1?

The answers to these questions are provided by Table 3-2. The left column lists the particular launch vehicles selected for this summary. The next four columns list, respectively, 1) the mission (by number) which the vehicle is capable of performing, 2) the total number of missions performed, 3) the total flight time involved, and, 4) the average flight time per mission. The last three columns contain numbers which are to be used in the comparison statement at the bottom of the Table.

Considering the first question a) of the comparison, it can be seen that the seven-segment (IIIF) Titan increases the number of missions performed from 4 to 7. It is also capable of decreasing the total required flight time of the five-segment (3D) Titan mission by 20 percent.

With regards to question b) of the comparison, it is obvious that something more than the Titan IIIF/Centaur/Burner II is needed to complete the program of selected missions. The last three vehicles given in Table 3-2 are presented as possibilities. A solar-electric stage addition to the Titan IIIF Centaur performs all 12 missions (satellite Callisto only for mission No. 9), with a 16 percent flight time decrease over the Titan IIIF/Centaur/ Burner II missions. Similar comparisons are apparent in the table for the addition of a hydrogen-fluorine Kick stage, or advancing to the Intermediate SIC/SIVB/Centaur launch vehicle. Note that the Kick stage addition cannot perform any of the Galilean satellite lander missions and the Saturn vehicle can also only perform the satellite lander mission (No. 9) at Callisto.

A number of conclusions were drawn from these results. The seven-segment solids improve launch vehicle capability for outer planets exploration. There seems to be little or no flight time advantage from the Burner II stage on

TABLE 3-2

SUMMARY COMPARISONS OF

CANDIDATE OUTER PLANET MISSIONS LAUNCH VEHICLES

Launch Vehicle	Possible Missions (Nos.)	Total Missions	Total Time (Years)	Average Mission Time (Yrs.)	Comparison*		
A					B	C	D
TITAN IIID-CENT-BII	1,2,3,4	4	23.1	5.78	--	-	--
TITAN IIIF-CENT-BII	1-4,6,7,12	7	29.8	4.26	20	4	TITAN IIID-CENT-BII
TITAN IIIF-CENT-SEP	1-12	12	61.5	5.13	16	7	TITAN IIIF-CENT-BII
TITAN IIIF-CENT I-KICK	1-8, 10-12	11	58.5	5.32	18	7	TITAN IIIF-CENT-BII
SIC-SIVB-CENTAUR	1-12	12	54.7	4.56	22	7	TITAN IIIF-CENT-BII

* To make comparisons between vehicles insert values from Columns A, B, C, and D into the following sentence:

"The (Column A) will provide a (Column B) % improvement in the total flight time of the (Column C) missions which can be performed by the (Column D) launch vehicle."

the seven-segment Titan. Something more than the Titan IIIF Centaur will be needed to carry outer planet exploration into the 1980's. The solar-electric and high-energy Kick stages are roughly equivalent, although there is an inclination to favor solar-electric propulsion because it can do the Callisto lander mission and may be more versatile for other applications, e.g., Mercury orbiters, comet rendezvous missions, high-data orbiters, etc. Using an Intermediate Saturn-class vehicle does not seem like a good solution to shorter flight times (for 1500 lbs. orbiters) since the orbit retro stages become enormous, from 5 to 10 times the weight of the orbiting spacecraft. A better answer to shorter flight times, particularly for Uranus and Neptune orbiters would be the introduction of the nuclear-electric low-thrust stage. It also would be desirable for the Galilean satellite lander mission.

Only launch vehicle comparisons have been made here. The effects of different retro stages, different orbits and different spacecraft weights also need to be considered.

FUTURE UPPER STAGE PROPULSION REQUIREMENTS

John C. Niehoff

This memo responds to a directive[1] from Robert S. Kraemer concerning upper stage propulsion requirements. Specifically considered are the launch vehicle (interplanetary injection) and retro stage (orbit capture) propulsion requirements for Jupiter and Saturn orbiter missions through 1988. The candidate launch systems reviewed to meet these requirements are restricted to derivations of the Titan IIID/Centaur vehicle. (The conclusions of a previous launch vehicle comparison study[2] and an estimated cost increase of $60M/copy were considered sufficient reasons not to include the Intermediate-20 (Saturn class) vehicle at this time).

Ballistic interplanetary trajectories to Jupiter and Saturn were surveyed for high-thrust stage launch vehicle combinations. Jupiter opportunities for the period 1974-85 with a flight time of 760 days (~ 2 yr.) and Saturn opportunities for the period 1980-85 with flight times of 4 and 5 years were considered. For payload performance analysis a maximum V_c of 48,000 ft/sec was selected for Jupiter transfers which eliminated the 1978 and 1985 opportunities and shortened the 1979 launch window to slightly less than the nominal 10 days. For Saturn transfers a maximum V_c of 52,500 ft/sec was assumed, which shortened the launch window of the 1980, 5-year (flight time) opportunity.

---

1 "Future Upper-Stage Propulsion Requirements", Kraemer, R. S., Planetary Programs (SL) Memo, NASA Headquarters, April 4, 1970.

2 "Outer-Planet Mission Justification for the Seven-Segment (1207) Titan Launch Vehicle", Niehoff, J. C., Astro Sciences Memo, IIT Research Institute, January 7, 1970.

Low-thrust interplanetary transfers were also considered for Jupiter and Saturn orbiter missions using a solar electric (SEP) stage combined with Titan/Centaur launch vehicles. Two dimensional trajectory data by Horsewood and Mann was evaluated to determine the appropriate interplanetary transfer and arrival conditions. The solar-electric trajectories selected were matched in hyperbolic approach speed, VHP, to the maximum values for the ballistic Jupiter and Saturn transfers so that subsequent retro stage computations would be equally applicable to either transfer mode. The chosen trajectories and payload capabilities are:

	VHP (km/sec)	Flight Time	Interplanetary Payloads (lbs)	
			Titan IIID/Cent	Titan IIID(7)/Cent
Jupiter	7.28	740^d	3450	4890
	7.93	3.78^y	2520	3600
Saturn	5.90	4.68^y	2760	3930

The flight times shown here are slightly shorter than for the ballistic cases (760^d, 4^y, 5^y). The payloads are reduced 20% from optimum to approximate off-optimum power (15 kw) and specific impulse (3500 sec). The primary difference between the low-thrust and ballistic transfer modes considered is the payload available for the retro propulsion system.

Candidate orbits considered in this analysis for total Jupiter and Saturn orbiter missions were constrained to periapses of 3 planet radii and orbital periods of 15, 30, 45 and 60 days at Jupiter and 20, 40, 60 and 80 days at Saturn, and were chosen to cover the range of possible interest. Capture impulses for insertion into inclined orbits (one impulse) and equatorial orbits (three impulse) were calculated to size the retro requirements. There is little difference in total impulse required between the 45 and 60 day orbits at Jupiter and between the 60 and 80 day orbits at Saturn, so the 60 day Jupiter and 80 day Saturn orbits were dropped from further consideration. The largest single capture impulses for inclined orbits at Jupiter occur in 1980 and are 1.83, 1.43 and 1.27 km/sec for the 15, 30 and 45 day orbits, respectively. For equatorial orbits the largest total impulses occur for both the 1983 and 1984 opportunities and are 2.07, 1.67 and 1.52 km/sec for the 15, 20 and 45 day orbits, respectively. This data was used to size the retro stages for the Jupiter orbiter payload because by using the most severe mission requirements the retro stages are automatically applicable to any of the other opportunities considered in this memo. Similar capture impulse data was analyzed for the Saturn orbits (where the 1985 opportunity has the highest impulse requirements).

There are several desirable characteristics which retro propulsion systems for outer planet orbiters should have. Multi-restart capability is essential if a single propulsion system is used to perform the one to three orbit insertion burns and several interplanetary midcourse maneuvers. The propellant should have a high specific impulse to minimize the retro system weight (and allow more science-oriented payload to be included in the spacecraft). The retro acceleration should also be kept below 0.5 g's in order not to structurally damage the spacecraft. In this study a 1500 lb spacecraft was used. With another 1500 lb added for the retro propulsion system, the retro engine thrust level should be about 600 lb (0.2 g's). The retro systems considered were:

fluoride-diborane (OF_2/B_2H_6), $I_{sp} \cong 400$ sec

fluroine-hydrazine (F_2/N_2H_4), $I_{sp} \cong 375$ sec

flox-methane ($FLOX/CH_4$), $I_{sp} \cong 365$ sec

beryllumininized solid, $I_{sp} \cong 315$ sec.

In order to limit the number of propulsion alternatives for payload analysis the OF_2/B_2H_6 and F_2/N_2H_4 combinations were considered equivalent with an I_{sp} of 390 sec. The FLOX/CH_4 propellant combination was dropped from further consideration because of its comparatively low I_{sp}. The berylliminized solid was considered at an I_{sp} of 315 sec.

PERFORMANCE RESULTS

A payload analysis was performed for 12 different launch vehicle-retro propulsion combinations to determine how many of the orbit options each combination could attain with a 1500 lb. spacecraft. The six launch vehicles considered were:

Titan IIID/Centaur/Burner II
Titan IIID(7)/Centaur/Burner II
Titan IIID/Centaur/HFK(10K)[3]
Titan IIID/(7)/Centaur/HFK(15K)[3]
Titan IIID/Centaur/SEP
Titan IIID(7)/Centaur/SEP

Adding the solid or the space-storable (S/S) retro stage gives the 12 launch vehicle-retro propulsion system.

The results of the payload analysis are presented as a bar chart in Figure 3-1. The various mission combinations are shown on the left side of the figure increasing in energy (propulsion) requirement from bottom to top. The launch vehicle/ retro propulsion combinations are given at the bottom of the figure, increasing from left to right in energy (propulsion) capability. The number of missions which can be performed by any specific propulsion combination is indicated by the height

---

[3] The number in parenthesis indicate the propellant loading in thousands of pounds for the Hydrogen-Flourine Kick stage (HFK).

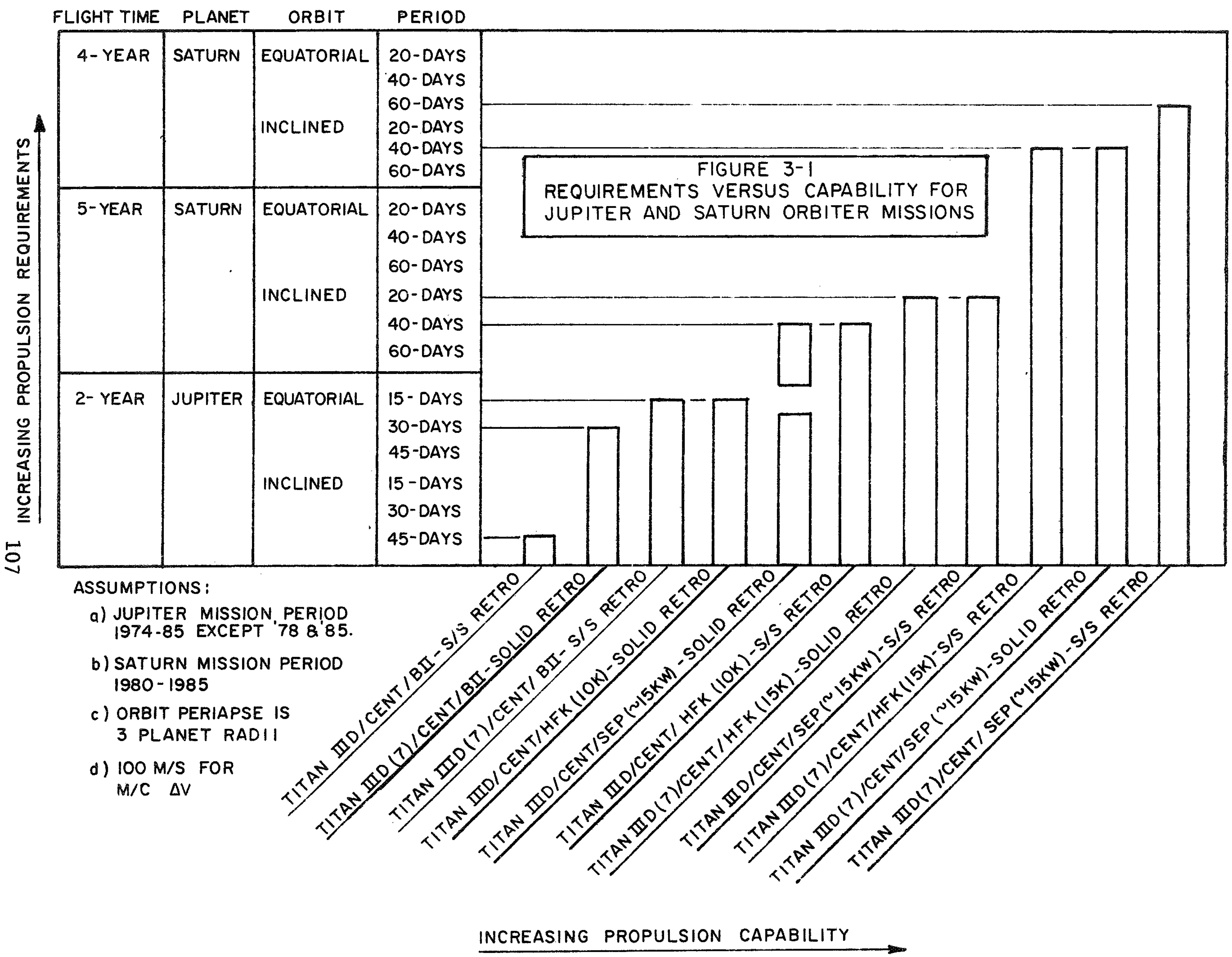


FIGURE 3-1
REQUIREMENTS VERSUS CAPABILITY FOR JUPITER AND SATURN ORBITER MISSIONS

of the bar above it. The only instance where propulsion requirements did not match the ordering of Figure 3-1 is for vehicle combination number 6, the Titan IIID/Centaur/SEP-Solid Retro, which cannot perform the 15-day Jupiter equatorial but can do the 60 and 40-day inclined orbit 5-year missions to Saturn.

It is also interesting to note that the most energetic vehicle combination, the Titan IIID(7)/Centaur/SEP-S/S Retro, can just attain equatorial orbits at Saturn if flight time is reduced to 4 years. This provided further confirmation to the conclusion from past mission studies that 4 years seems to be a lower flight time limit for Saturn orbiter missions.

None of the 12 combinations shown in Figure 3-1 are being actively developed today for a specific flight program. It is obvious that one of these or some other competing propulsion combination must be chosen and built if there are going to be 1500 lb or heavier outer planet orbiters. It is also apparent that NASA is not likely to invest in more than one or two propulsion developments concurrently. This memo provides eight different development sequences for the propulsion combinations considered. The development sequence providing the slowest capability improvement and the lowest ultimate payload is:

a) Titan IIID/Centaur/Burner II - Solid Retro
b) Titan IIID(7)/Centaur/Burner II - Solid Retro
c) Titan IIID(7)/Centaur/HFK(15K) - Solid Retro.

The fastest improvement sequence leading to the best ultimate payload capability is:

a) Titan IIID/Centaur/Burner II - S/S Retro
b) Titan IIID/Centaur/SEP(15kw) - S/S Retro
c) Titan IIID(7)/Centaur/SEP(15kw) - S/S Retro.

CONCLUSIONS

The development of a retro stage would appear to hinge more on system flexibility, reliability, and cost than on performance, provided that consideration is limited to OF_2/B_2H_6, F_2/N_2H_4, berylliminized solids or other propellant combinations of comparable performance (including stage hardware weight as well as I_{sp}). The F_2/N_2H_4 system described by D. Dipprey[4] best meets these criteria, although more development work needs to be done with it before this can be a firm conclusion.

A conclusion regarding launch vehicle development is much more difficult, if not impossible, at this time. The uncertainty in cost of competing systems tends to turn the problem into a "political football". In addition to those combinations considered here the Intermediate-20 and the Shuttle must be recognized as competing capabilities. The Intermediate-20 introduces an element of over-kill for Jupiter and Saturn missions, but this is countered by broader applications (Uranus and Neptune missions) and the possibility of multi-mission (dual spacecraft) launches. Performance data for the Shuttle probably has not been sufficiently stabilized yet to determine its status of competitiveness. If these important factors are momentarily disregarded and just Titan-class vehicles are considered, then it is concluded that the addition of a solar-electric low-thrust stage is the single most useful improvement that can be made in the capability of the Titan IIID/Centaur/Burner II.

---

[4] April 24, 1970 letter to J. Salmanson, from D. F. Dipprey, Manager, Liquid Propulsion Section, Jet Propulsion Laboratory.

3.2 COMPUTER CODES DEVELOPED

The following computer codes were developed for use on several of the studies performed under Contract NASW-2023 and added to the Astro Sciences program inventory within the last year.

PETARD: Similar to "KOFNAL". Generates ground traces of orbiting spacecraft for any number of desired revolutions for any of the nine planets of the solar system. Has Calcomp capability for plotting latitude or altitude as a function of time from periapse on semi-log plots.

CAPTR: Set of two codes developed to perform orbit and landing maneuvers about a natural planetary satellite.

ETY 1: Solves differential equations describing motion of a spacecraft entering the atmosphere of a rotating planet with a spherical gravity field. Present version assumes fixed values of the drag coefficient and lift to drag ratio. Atmospheric density is computed as an exponential function of altitude.

4. PAPERS PRESENTED AND PUBLISHED

"TOURING THE GALILEAN SATELLITES"
By J. C. Niehoff
Presented at the AAS/AIAA Astrodynamics Conference, Santa Barbara, California, August 19-21, 1970
AIAA Paper No. 70-1070
To be published.

ABSTRACT

An interesting technique is presented for exploiting orbiting Jupiter spacecraft to repeatedly encounter the Galilean satellites. Commensurable orbits are derived, assuming coplanar two-body motion, which mesh with the motion of Io, Europa and Ganymede to provide multiple satellite flybys. A practical 14-day orbit is presented for the 1981 Jupiter opportunity which provides 35 satellite encounters, including several with Callisto, over a period of 170 days. Encounter orientation and lighting conditions are favorable for surface imagery. Satellite gravitational perturbations are sufficient to upset the predicted encounter sequences. An inexpensive impulse policy is presented which provides the needed orbit control.

"MISSIONS TO MERCURY (1973-1990)"
By D. A. Klopp, D. L. Roberts, and W. C. Wells
Presented at the 16th Annual Meeting, American Astronautical Society, Anaheim, California, June 1970.
To be published.

ABSTRACT

This paper discusses the scientific objectives of space missions to Mercury and presents representative science payloads for flyby and orbiter missions. Ballistic mission modes (both direct and Venus swingby) and low-thrust mission modes (solar-electric) are considered and interpreted in terms of launch vehicle or payload capability. Problems in achieving suitable coverage and illumination conditions are emphasized. Flyby missions are not likely to provide complete coverage of Mercury, while approximately 180 days in orbit are required to obtain the visual and thermal mapping desired. An unmanned Mercury orbiter mission is comparable to Apollo 8, in terms of complexity and energy requirements.

"TRAJECTORY REQUIREMENTS FOR COMET RENDEZVOUS"

By A. L. Friedlander, J. C. Niehoff, and
J. I. Waters

Presented at the AAS/AIAA Astrodynamics Conference,
Santa Barbara, California, August 19-21, 1970.
AIAA Paper No. 70-1072.
To be published.

ABSTRACT

This paper presents a new look at spacecraft mission opportunities to the short-period comets in the time period 1975-95. The objective is to identify the most promising rendezvous opportunities and flight modes from the standpoint of trajectory requirements and launch vehicle/payload capabilities. A "broad-bush" treatment of wide scope underlies the analysis. Selection criteria leading to 16 comet apparitions for study are described. The candidate flight modes include; 3-impulse ballistic transfers, Jupiter gravity-assist transfers, solar-electric and nuclear-electric low thrust transfers. Results show that among the best early opportunities are comets Encke/80, d'Arrest/82 and Kopff/83. Although these missions can be performed ballistically, solar-electric propulsion offers greatly improved performance. Practical accomplishment of the very difficult Halley rendezvous depends upon the development and availability of nuclear-electric propulsion by 1983.

"APPLICATION OF AN IMPULSIVE TRAJECTORY OPTIMIZATION METHOD TO THE COMET RENDEZVOUS PROBLEM"

By J. I. Waters

To be presented at the Ninth Aerospace Sciences Meeting, American Institute of Aeronautics and Astronautics, New York, New York, January 25-27, 1971

ABSTRACT

A multiple impulsive trajectory optimization technique has been applied to the comet rendezvous problem. The resulting computer program employs the conjugate gradient search method to find minimum ΔV trajectories subject to initial and final position constraints and automatically inserts additional impulses along the trajectory as indicated by examination of the primer vector. Optimum impulsive rendezvous trajectories to the short period comets fall into two distinct categories requiring three and five impulses respectively. The three impulse class is characterized by a small angle between the nodal and apsidal lines which allows the plane change and gross energy adjustment to be combined.

5. BIBLIOGRAPHY OF AS/IITRI REPORTS AND TECHNICAL MEMORANDA

5. BIBLIOGRAPHY OF AS/IITRI REPORTS AND TECHNICAL MEMORANDA

The following bibliography contains all those reports and technical memoranda published by Astro Sciences.

TM C-3 An Empirical Approach to Estimating Space Program Costs, by J. Beverly, C. Stone and R. Vickers (copies not available).

R C-4 Progress on Spacecraft Cost Estimation Studies, by J. Beverly and C. Stone (copies not available).

TM C-5 An Analysis of the Correlation Between Spacecraft Performance and Cost Complexity Factor, by W. Finnegan (copies not available).

R C-6 Spacecraft Cost Estimation, by W. Finnegan and C. Stone, NASA STAR No. N66-29740.

R C-7 Spacecraft Program Cost Estimating Manual, by W. Finnegan and C. Stone, NASA STAR No. N66-30762.

R C-8 Spacecraft Comparison Study for Mars-Venus Fields and Particles Orbiters, by W. Adams and H. Goldman.

R M-1 Survey of a Jovian Mission, by ASC Staff, NASA STAR No. N64-20643.

R M-2 Survey of a Jovian Mission (U), Confidential (copies not available).

R M-3 Survey of Missions to the Asteroids, by A. Friedlander and R. Vickers, NASA STAR No. N64-19566.

R M-4 Summary of Flight Missions to Jupiter, by ASC Staff, NASA STAR No. N64-26597.

R M-5 Missions to the Asteroids, by ASC Staff (copies not available).

R M-6 A study of Interplanetary Space Missions, by D. L. Roberts, NASA STAR No. N65-25003.

R M-7 A Survey of Comet Missions, by D. L. Roberts, NASA STAR No. N65-30481.

TM M-8 Cometary Study by Means of Space Missions, by F. Narin, P. Pierce and D. L. Roberts (copies not available).

R M-9 Missions to the Comets, by F. Narin, P. Pierce and D. L. Roberts, NASA STAR No. N66-15978.

TM M-10 The Satellites of Mars, by D. L. Roberts (copies not available).

R M-11 A Survey of Missions to Saturn, Uranus, Neptune and Pluto, by F. Narin et al., NASA STAR No. N67-14253.

R M-12 A Survey of Multiple Missions Using Gravity-Assisted Trajectories, by J. C. Niehoff, NASA STAR No.N66-32440.

R M-13 Preliminary Payload Analysis of Automated Mars Sample Return Missions, by J. C. Niehoff, NASA STAR No. N67-28833.

R M-14 Digest Report: Missions to the Outer Planets, by F. Narin, NASA STAR No. N67-30917.

TM M-15 A Solar System Total Exploration Planning System (STEPS), by J. Witting.

TM M-16 The Multiple Outer Planet Mission (Grand Tour), by ASC Staff.

TM M-17 A Preliminary Study of Jupiter Atmospheric Missions, by J. E. Gilligan and D. L. Roberts.

TM M-18 Contribution to the Lunar Program Memorandum 1969, by D. L. Roberts.

R M-19 Preliminary Feasibility Study of Soft Lander Missions to the Galilean Satellites of Jupiter, by M. J. Price and D. J. Spadoni.

R M-20 Jupiter Orbiter Mission Study, by J. C. Niehoff.

R M-21 Solar Electric Propulsion - A Survey, by A. Friedlander.

TM M-22 A Preliminary Comparison of Direct Flybys and Swingby Tours of the Outer Planets, by A. Friedlander and R. Brandenburg.

TM M-23 Lunar Exploration Program Memorandum 1970-71, by D. L. Roberts.

R M-24 Solar Electric Propulsion Applications for Jupiter and Saturn Orbiter Missions, by A. Friedlander and R. Brandenburg.

R M-25 A Preliminary Feasibility Study of Composite Orbiter/ Lander Missions to the Satellites of the Outer Planets, by M. J. Price and D. J. Spadoni.

R P-1 Scientific Objectives of Deep Space Investigations - Jupiter, by D. L. Roberts, NASA STAR No. 64-19467.

R P-2 Scientific Objectives of Deep Space Investigations - The Satellites of Jupiter, by D. L. Roberts, NASA STAR No. N64-19568.

R P-3 Scientific Objectives of Deep Space Investigations - Comets, by D. L. Roberts, NASA STAR No. N64-19569.

R P-4 Scientific Objectives of Deep Space Investigations - Asteroids, by D. L. Roberts, NASA STAR No. N64-19579.

R P-5 Scientific Objectives of Deep Space Investigations - Interplanetary Space Beyond 1 AU, by D. L. Roberts, NASA STAR No. N64-19471.

R P-6 Scientific Objectives for Mercury Missions, by T. Owen, NASA STAR No. N64-26599.

R P-7 Scientific Objectives for Deep Space Investigations - Venus, by P. J. Dickerman, NASA STAR No. N66-32439.

R P-8 Scientific Objectives of Deep Space Investigations - Non-Ecliptic Regions, by D. L. Roberts (copies not available).

R P-9 Copendium of Data on Some Periodic Comets, by D. L. Roberts, NASA STAR No. N64-28524.

R P-10 Critical Measurements on Early Missions to Jupiter, by J. Whtting, M. W. P. Cann, and T. Owen, NASA STAR No. N66-15807.

R P-11 Scientific Objectives of Deep Space Investigations - Saturn, Uranus, Neptune and Pluto, by P. J. Dickerman, NASA STAR No. N66-17090.

TM P-12 Regularities in the Solar System Pertaining to Its Origin and Evolution, by J. Witting, (copies not available).

TM P-13 Comparison Criteria for a Total Lunar Scientific Exploration Program Study, by C. A.Stone, (copies not available).

R P-14 Analytical Methods and Observational Requirements for Interpretations of Asteroid Distributions, by J. Ash, NASA STAR No. N67-17961.

TM P-15 Analytical Techniques for the Investigation of Distributional Features of the Asteroids, by J. Ash (copies not available).

R P-16 Mission Requirements for Exobiological Measurements on Venus, by W. Riesen and D. L. Roberts, NASA STAR No. N67-12073.

TM P-17 A Geological Analysis for Lunar Exploration, by W. Scoggins.

R P-18 Scientific Objectives of Deep Space Investigations: The Origin and Evolution of the Solar System, by J. Witting, NASA STAR No. N67-10880.

R P-19 Scientific Objectives of Deep Space Investigations: Jupiter As An Object of Biological Interest, by ASC Staff, NASA STAR No. N67-27647.

R P-20 Suggested Measurement/Instrument Requirements for Lunar Orbiter Block III, by W. Scoggins and D. L. Roberts, NASA STAR No. N67-31059.

TM P-21 Scientific Objectives for Total Planetary Exploration, by ASC Staff.

TM P-22 Role of Ground Base Observations in the Exploration of Venus, by J. T. Dockery.

R P-23 A Preliminary Evaluation of the Applicability of Surface Sampling to Mars Exploration, by W. H. Scoggins and D. L. Roberts.

R P-24 The Scientific Objectives for Venus Landers, by J. Gilligan (copies not available).

TM P-25 Preliminary Study of Atmospheric Sample Return from Venus, by G. Woodman.

R P-26 Apollo G-1 Mission Science Data Dissemination Study, by D. L. Roberts et al.

R P-27	Scientific Objectives of Total Planetary Exploration, by J. C. Jones.
R P-28	Objective Priorities for Lunar Science Orbital Instruments, by W. Adams, A. Binder, H. Goldman, and W. Hartmann.
R P-29	Logic for Lunar Science Objectives, by A. Binder, W. Hartmann, D. L. Roberts, and R. Sullivan.
R P-30	Criteria for Lunar Site Selection, by D. L. Roberts and A. Binder.
R P-31	Lunar Surface Scientific Experiments and Emplaced Station Science, by W. Hartmann.
R P-32	Objectives of Permanent Lunar Bases, by W. Hartmann and R. Sullivan.
R P-33	The Role of Subsatellites in the Exploration of the Moon, by P. Dickerman.
R P-34	Candidate Lunar Experiments, by R. Sullivan et al.
R P-35	Apollo 18 and 19 Mission and Science Options, by J. Blahnik.
R P-36	The Physical Structure of the Saturn Ring System, by M. J. Price.
R P-37	Analysis of the Apollo II Results, by A. Binder and W. Hartmann.
TM R-1	Comparative Reliability Estimation Method for Mission Programming, by H. Lauffenberger (copies not available).
R R-2	Probability of Biological Contamination of Mars, by A. Ungar, R. Wheeler and D. L. Roberts (copies not available).

TM S-1 Study of Photographic and Spectrometric Subsystems for Voyager, by P. N. Slater and G. Johnson (copies not available).

R S-2 Scientific Questions Requiring Advanced Technology: Asteroid Fly-Through Mission, by J. A. Greenspan, NASA STAR No. N66-23631.

R S-3 Telemetry Communications Guideline, by M. Stein, (copies not available).

R S-4 Thermophysical Effects and Feasibility of Jupiter Atmospheric Entry, by J. E. Gilligan.

TM S-5 Low-Thrust and Ballistic Payload Comparison for Jupiter Orbiter Missions, by D. Healy and D. L. Roberts (copies not available).

TM S-6 Deep Space Communications: Command Link and Atmospheric Probe Entry, by M. S. Stein and D. L. Roberts.

TM S-7 Radar Exploration of Venus, by D. L. Roberts and H. Goldman.

R S-8 Planetary Exploration Potentials of Spacecraft Radar, by H. Goldman and R. Brandenburg.

R T-4R Summary To One-Way Ballistic Trajectory Data: Earth to Solar System Targets, by F. Narin and P. Pierce, NASA STAR No. N64-19572.

R T-5 Accuracy and Capabilities of ASC/IITRI Conic Section Trajectory System, by P. Pierce and F. Narin, NASA STAR No. N64-19603.

R T-6 Accessible Regions Method of Energy and Flight Time Analysis For One-Way Ballistic Interplanetary Missions, by F. Narin, NASA STAR No. N64-28840.

R T-7 Perturbations, Sighting and Trajectory Analysis for Periodic Comets: 1965-1975, by F. Narin and P. Pierce, NASA STAR No. N66-13398.

TM T-8 Comparison of Atlas Centaur and Floxed Atlas Centaur Capabilities in Interplanetary Exploration Using the Accessible Regions Method, by F. Narin, (copies not available).

R T-9 Spatial Distribution of the Known Asteroids, by F. Narin NASA STAR No. N65-30471.

TM T-10 Collected Launch Vehicle Curves, by F. Narin (copies not available).

R T-11 Sighting and Trajectory Analysis for Periodic Comets: 1975-1986, by F. Narin and B. Rejzer, NASA STAR No. N65-28347.

R T-12 Analysis of Gravity-Assisted Trajectories in the Ecliptic Plane, by J. C. Niehoff, NASA STAR No. N65-34460.

R T-13 Trajectory and Sighting Analysis for First Apparition Comets, by P. Pierce, NASA STAR No. N65-35845.

R T-14 Low-Thrust Trajectory and Payload Analysis for Solar System Exploration Utilizing the Accessible Regions Method, by A. Friedlander, NASA STAR No. N66-13992.

TM T-15 Mission Requirements for Unmanned Exploration of the Solar System, by F. Narin (copies not available).

TM T-16 Selection of Comet Missions: 1965-1986, by F. Narin, P. Pierce and D. L. Roberts (copies not available).

R T-17 Low-Thrust Trajectory Capabilities for Exploration of the Solar System, by A. Friedlander, NASA STAR No. N67-12224.

R T-18 The Accessible Regions Presentation of Gravity-Assisted Trajectories Using Jupiter, by D. A. Klopp and J. C. Niehoff, NASA STAR No. N67-34604.

R T-19 On The Problem of Comet Orbit Determination for Spacecraft Intercept Missions, by A. Friedlander, NASA STAR No. 67-28832.

R T-20 Trajectory Opportunities to the Outer Planets for the Period 1975-2000 (BR).

TM T-21 Comet Rendezvous Opportunities (An Interim Report), by A. Friedlander, J. C. Niehoff, and J. I. Waters.

R T-22 Mars Orbit Characteristics Related to Experiment Design - D. L. Roberts and M. Hopper.

TM T-23 Gravity-Assist/Low Thrust Hybrid Systems, by J. I. Waters.

TM T-24 Analytical Solution of Low-Thrust Trajectories by Time Series Approximation of Acceleration Functions, by A. Friedlander.

R T-25 Trajectory and Propulsion Characteristics of Comet Rendezvous Opportunities, by A. Friedlander, J. C. Niehoff, and J. I. Waters.

TM T-26 Preliminary Report on Apollo Site Selection, by W. Hartmann.

TM-1 Notes on the Lunar Atmosphere, by W. O. Davies.

TM-2 Comments on the Experimental Objectives of The A.E.S. Program, (2) HF-VHF Reflectivity, by H. J. Goldman.

TM-3 Examination of the Lunar Surface by Solar X-Ray Fluorescence, by E. Thornton.

TM-4 "Conic Section Trajectories: Summary of the Solar System, by F. Narin.

TM-5 Comments on the Experimental Objectives of The A.E.S. Program, (5) Radar Imaging, by H. J. Goldman.

TM-6 Survey of Power Systems for Early Lunar "Stay-Behind" Experiments, by G. Walker.

TM-7 Ultraviolet Reflectance and Ultraviolet Stimulated Luminescence of the Earth's Surface, by P. Dickerman.

TM-8 Radiation and Micrometeorite Environmental Hazards to Apollo, by T. Stinchomb and R. L. Chandler.

TM-9 Radiation Effects on Films in Synchronous Earth Orbit Missions, by T. Stinchomb and H. Watts.

TM-11 Power Systems for the Lunar Surface Experimental Package, by G. Walker.

TM-12 Preliminary Geological Analysis of Lunar Orbital Sensors, by B. Pauling and R. Robson.

TM-13 Checkout of Apollo Application Program Experiments, by H. R. Hegner.

TM-14 Non-Imaging Infrared Instrument Parametric Study, by H. T. Betz and M. S. Stein.

TM-15 Preliminary Summary of Manned Mission Support Requirements for Space Science and Applications Objectives, by J. G. Barmby and R. G. Dubinsky.

TM-16 Preliminary Analysis of Spacecraft Commonality for the Space Applications Program (1970-1986), by J. G. Barmby, J. E. Orth, and W. L. Vest.

TM-17 A Method for Determining Optimum Experiment Profiles and Resultant Data Bulk Requirements for Remote Imaging of the Lunar Surface From Polar Orbit, by P. Bock.

TM-18 Scientific Experiment Program for Earth-Orbital Flights of Manned Spacecraft, by R. G. Dubinsky.

TM-19 Determination of Earth Orbital Experiment Profiles and Data Requirements, by P. Bock.

TM-21 Optical Imagers for the Small Earth Resources Satellite, by S. S. Verner.

TM-22 Compendium of Space Applications Sensors and Instruments, by J. E. Orth.

TM-23 Basic Data for Earth Resources Survey Program Map Plan, by K. Clark.

TM-26 Experiment Profile Analysis of the Multiband Camera Sun Synchronous Mission, by P. Bock and H. Lane.

6. MAJOR COMPUTATIONAL CODES

6. MAJOR COMPUTATIONAL CODES

The following computer codes have been written or adapted for use on contract studies between March 1963 and October 1970.

INTERPLANETARY TRANSFERS

Conic Section Codes

SPARC: The JPL general conic section code for ballistic and ballistic-gravity-assist flights.

ASC CONIC: An extensive collection of programs and subprograms for ballistic and gravity-assist flights and accessible regions calculations, and for conic guidance analysis.

TOPSY: Determines the minimum ideal velocity and the corresponding time required to reach any point in the solar system.

High Precision Codes

NBODY (II): The Fortran II version of the Lewis Research Center code has been used for comet perturbation analysis, considering the gravitational effects of Sun and planets simultaneously.

NBODY (IV): The Fortran IV version of this has been revised at ASC for multibody, high precision targeting and guidance analysis.

Low Thrust Codes

JPL CODE: The JPL Calculus of Variations Optimized Thrusted Trajectory Code has been used for optimum interplanetary nuclear electric flight with variable thrust, constant thrust, or constant acceleration.

UNITED AIRCRAFT CODE: Computes optimum low thrust (nuclear-electric) interplanetary trajectories under constant thrust conditions. Method employed is calculus of variations and finite difference Newton-Raphson Algorithm. Powerplant mass fraction and specific impulse can be optimized if desired.

BOEING CODE: CHEBYTOP is a fast generator of optimum low thrust interplanetary trajectories. Both solar-electric and nuclear electric powerplants can be treated. Propulsion system parameters must be specified - payload optimization can be accomplished by multiple parametric runs.

MULIMP: Uses Conjugate-Gradient search method to find minimum, ΔV trajectories consisting of up to four free fall conic arcs separated by up to five impulses. Departure is from Earth orbit and the arrival point is constrained to lie on an arbitrary conic. Velocity is matched at the arrival point (rendezvous).

Near Planet Operations

ATMENT: One of a series of codes for integrating the atmospheric entry for a spacecraft.

ZAYIN: A Fortran II code (from W. P. Overbeck) modified for calculating satellite orbits around the Earth, including oblateness and air drag.

GNDTRC: Generates lunar ground traces for specified lunar orbits.

LIMITS: Computes maximum velocity and maximum energy change as a function of miss distance from a given gravity-assist body.

KOFNAL: Generates ground traces of orbiting spacecraft for any number of desired revolutions. Can be used for all nine planets of the solar system. Has Calcomp capability for plotting longitude and latitude of the ground trace.

CONTUR: Generates data for Sun, Earth & Canopus occultation contours for hyperbolic flybys past any given planet.

AMSOCC: Generates data for Sun, Earth & Canopus occultation contours for orbiting spacecraft about any given planet.

HYPTRC: Computes 2-D planetary encounter trajectories in polar coordinates given heliocentric transfer trajectory from Earth.

TRACE: Generates Earth ground traces for specified Earth orbits.

PROFYL: A planetary encounter profile definition code.

RINGER: A code of calculating crossings of Saturn's ring plane during flyby.

Guidance and Orbit Determination

ORBDET: Orbit determination for an overdetermined set of points by Kalman filtering.

LTNAV: A low thrust navigation code.

PARODE: A radio tracking performance evaluation code for orbit determination during planetary approach.

COMODE: High precision comet orbit determination code, taking into consideration gravitational effects of Sun and all nine planets simultaneously.

ORBOBS: A Fortran IV program for determining minimum separation intercepts of a Jupiter orbiter with the four Galilean Satellites; Io, Europa, Ganymede, and Callisto.

CELESTIAL TRACKING: A celestial tracking performance evaluation code for orbit determination during planetary approach.

SURVEY: Generates sighting conditions for comets over a specified length of time. Has Calcomp capability for plotting sighting conditions as function of time from perihelion.

Combinatorial Codes

XPSLCT and COMBSC - find various sets of payloads from experiments and instruments, subject to spacecraft constraints.

HFIT: A code for least square fit of a set of points to a hyperbola.

BIMED: A general statistical analysis package from UCLA used for multiple regression analysis.

IMP 3: An integer programming code.

Space Sciences Codes

ASTA: A set of codes for analyzing spatial and velocity distributions of the asteroids.

HAZARD: A code for calculating spacecraft to asteroid and meteor stream distances.

SIGHT: A code for analyzing positions of celestial objects.

INTEGRALS: A set of codes for evaluating various special integrals which arise in planetary atmosphere analysis.

Special Features and Systems

GPSS-III: An IBM system for analyses of systems of discrete transactions.

MIMIC: A Fortran IV-like system for simulating, on the 7094, an analog computer and thereby easily doing integrations.

KWIC-II: The IBM key word in context system used to catalog the ASC library of some 8000 documents.

ORBITAL ELEMENTS TAPE: An extensive collection of orbital elements for solar system objects, including planets, 1600 numbered asteroids, 2000 unnumbered asteroids and hundreds of comets.